RESEARCH MEMORANDUM

A SURVEY OF METHODS FOR TURBOJET THRUST MEASUREMENT

APPLICABLE TO FLIGHT INSTALLATIONS

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FOR AERONAUTICS
WASHINGTON

June 10, 1955
Declassified October 31, 1958

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SUMMARY

Experience accumulated during the past 10 years in measuring turbojet-engine thrust in altitude facilities is summarized herein to indicate the currently best-known techniques applicable for obtaining in-flight thrust measurements. Problems that are encountered in obtaining accurate thrust measurements are illustrated, and the resultant techniques required for measurement of the net thrust of several typical types of turbojet installations are discussed. Techniques for obtaining a performance calibration of the basic engine while it is installed in an airplane are also indicated.

This summary of turbojet thrust-measuring techniques indicates that, although there are no simple methods or short cuts to adequately substitute for extensive instrumentation, measurements of in-flight performance can be obtained with an adequate degree of accuracy by utilizing the instrumentation techniques discussed herein.

INTRODUCTION

Two of the most important power-plant characteristics that must be measured in flight are the thrust of the over-all installation and the performance of the basic engine. The problem of obtaining accurate performance measurements of a turbojet installation in flight is therefore one of considerable importance and has received the continued attention of the aircraft industry since the advent of turbojet aircraft. Determination of the thrust of the over-all installation in prototype or experimental aircraft is required in order to obtain the airplane drag coefficients. Performance of the basic engine is often necessary or desirable in order to determine whether the installed engine has retained the performance specified by the manufacturer. With the advent of transonic and supersonic turbojet aircraft, the problem of obtaining these measurements accurately assumes even greater importance, because of the

small margin between thrust and drag at supersonic speeds. At the same time, accurate thrust measurements have become even more difficult to obtain in supersonic aircraft because an afterburner with its ejector and other complicating components are usually involved.

Methods for accurately measuring turbojet thrust have been under continuous development in the altitude facilities at the NACA Lewis laboratory and at other research establishments during the past 10 years (refs. 1 to 18). From this experience, as well as a limited amount of flight-test experience of the NACA and other organizations, the currently best-known methods for obtaining in-flight turbojet thrust measurements and performance calibrations are summarized herein.

This information provides specific answers to the often-asked questions about the techniques that must be used and the complications in instrumentation that are required in order to obtain accurate turbojet thrust measurements in flight. The techniques suggested may not be applicable to many production aircraft because of space limitations, but rather are intended primarily for use in experimental or prototype aircraft in which a greater amount of instrumentation can be tolerated.

Because turbojet installations of varying degrees of complexity are used in current aircraft and the thrust-measuring techniques differ to some degree with each type of installation, five types of turbojet installations are considered in the present discussion. These installations, which are illustrated schematically in figure 1, are as follows:

(1) Fixed exhaust nozzle, no afterburner, no ejector

(2) Variable exhaust nozzle, no afterburner, no ejector

(3) Variable exhaust nozzle, with afterburner, no ejector

(4) Variable exhaust nozzle, no afterburner, with ejector

(5) Variable exhaust nozzle, with afterburner, with ejector.

Methods for obtaining the thrust of each type of installation are discussed, and the recommended instrumentation techniques are illustrated. The probable error in measuring the thrust of these installations is also indicated.

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
C_F	thrust coefficient
C_V	velocity coefficient
c_p	specific heat at constant pressure
F_j	jet thrust, lb
F_n	net thrust, lb
g	acceleration due to gravity, 32.2 ft/sec^2
H	enthalpy, Btu/lb
h_f	lower heating value of fuel, Btu/lb
M	Mach number
m	mass flow, slugs/sec
N	engine speed, rpm
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	gas constant, ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
V	velocity, ft/sec
W	weight flow, lb/sec
γ	ratio of specific heats
δ	ratio of engine-inlet total pressure to NACA standard sea-level pressure, $P_1/2116$
η	efficiency
θ	ratio of engine-inlet total temperature to NACA standard sea-level temperature, $T_1/519$

ρ density, $(lb)(sec)^2/ft^4$

φ ratio of viscosity of air to viscosity of air at NACA standard sea-level conditions

Subscripts:

a air

bl compressor-outlet bleed

C compressor

c combustor

f fuel

g gas

j jet

p primary

s secondary

T turbine

th turbine throat

w ejector-shroud wall

0 free-stream conditions

1 engine or compressor inlet

2 turbine outlet

3 exhaust-nozzle exit (also refers to nozzle-inlet total-pressure and -temperature measurements)

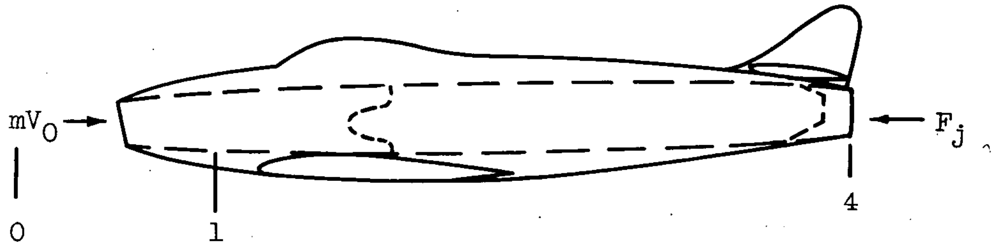
4 ejector-shroud exit or fuselage exit

METHOD OF ANALYSIS

Definition of Thrust

The net installation thrust with which this report is primarily concerned is defined in the conventional manner as the change in momentum of the gases passing through the engine installation. In its simplest form the net thrust is composed of two terms, the jet thrust and the inlet, or free-stream, momentum drag. For clarification, these terms are illustrated in sketch (a), where

$$F_n = F_j - mV_0 \tag{1}$$



(a) Forces acting on typical airplane.

In this discussion the jet thrust is defined as the change in momentum of all the gases (engine exhaust, cooling air, etc.) leaving the exit of the installation, designated station 4, and can be expressed as

$$F_j = m_4V_4 + A_4(p_4 - p_0) \tag{2}$$

The inlet-momentum term of equation (1) includes the momentum losses of all air quantities entering the installation, such as engine air and cooling air. This term may be expressed as

$$mV_0 = m_1V_0 + m_sV_0 \tag{3}$$

where m_1V_0 represents the free-stream momentum of the engine air and m_sV_0 represents the momentum of the boundary-layer air or cooling air, chargeable to the installation, that is taken aboard the airplane. Alternatively, part of the momentum drag of the boundary-layer air may be charged to the airplane by replacing m_sV_0 with the term m_sV_s, where V_s is the velocity immediately upstream of the secondary-air inlet.

Combining equations (2) and (3) in equation (1) results in the following expression for the net thrust of the installation:

$$F_n = \left[m_4V_4 + A_4(p_4 - p_0)\right] - m_1V_0 - m_sV_0 \tag{4}$$

By definition, all thrust losses due to cooling shrouds, or other internal losses, are included and any effects of external flow on the pressure in the plane of the installation exit are considered. External drag of the installation and interference effects of the inlet or exit on external drag are not considered in the thrust measurement.

Air-Flow Measurements

As shown in equation (4), a determination of engine-inlet air flow is necessary in order to permit calculation of the installation thrust for any type of engine installation. Because the methods of determining air flow are the same for all types of installations, air-flow measuring techniques are discussed first.

Two measuring stations have been commonly used in turbojet engines to determine air flow: (1) the engine inlet (station 1, fig. 1(a)) and (2) the choked turbine nozzle. The exhaust-nozzle exit has also been used (ref. 15), but the results have generally been less accurate than for the other two stations. This discussion is therefore limited to the use of the inlet and turbine-nozzle stations.

Equation for engine-inlet air flow. - At the engine inlet the equation for determining air flow is the simple one-dimensional continuity relation

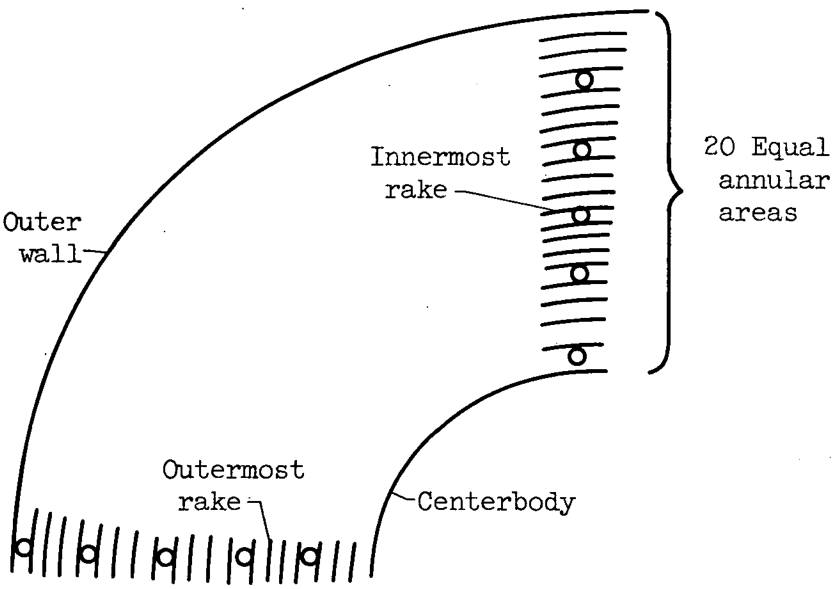
$$m_1 = \rho_1A_1V_1 = \frac{p_1A_1}{\sqrt{gRT_1}\left(\frac{p_1}{P_1}\right)^{\frac{\gamma_1-1}{\gamma_1}}}\sqrt{\frac{2\gamma_1}{\gamma_1 - 1}\left[1 - \left(\frac{p_1}{P_1}\right)^{\frac{\gamma_1-1}{\gamma_1}}\right]} \tag{5}$$

In order to determine the air flow at the engine inlet, instrumentation must therefore measure total and static pressure and total temperature. The area A_1 is, of course, the annular area at the station of measurement.

<u>Instrumentation for air-flow determination at engine inlet</u>. - The number of instruments required and the accuracy of the air-flow measurements at the engine inlet depend upon the pressure distribution at the engine inlet. About half the aircraft inlets for which pressure-distribution data are available produce engine-inlet total-pressure distributions that vary by more than ±5 percent. Experience in measuring engine-inlet air flow at the NACA has been predominantly in installations in which the total-pressure gradients at the inlet were within ±3 percent. Of course, the larger the pressure gradient at the inlet the greater the number of instruments required for an accurate determination of air flow. In one case in which a 20-percent variation in total pressure existed around the circumference at the inlet, the error in measured air flow was approximately $\pm 2\frac{1}{2}$ percent when four rakes of five total-pressure tubes spaced 90° apart were used. Installation of a larger number of rakes would, of course, reduce the error, but adequate data are not available to provide general rules for the number of rakes required with various pressure distributions. In the absence of more complete data, experiences with measurement of flow with nearly uniform inlet pressures, which will be discussed, may serve as a guide for air-flow measurements in inlets with large pressure gradients. The method of determining air flow at the turbine nozzles is applicable in the presence of inlet pressure gradients and is discussed in the following section.

A schematic diagram of a total-pressure and -temperature rake and wall static taps for use at the engine inlet is shown in figure 2. Common practice has been to use six to eight total-pressure tubes per rake and two or three total-temperature thermocouples per rake. The total-pressure tubes nearest the wall should be spaced as near the walls as possible in order to pick up any boundary-layer effect present. The radial spacing of the remaining tubes is usually determined by dividing the annulus into equal annular areas equivalent in number to the total number of total-pressure tubes in the survey. Each rake is then radially offset a slight amount from each of the others so that one pressure

tube is located in each of the annular areas, as illustrated by sketch (b).



(b) Arrangement for four rakes with five total-pressure tubes each.

The survey plane should always be located at least 6 inches upstream of any struts or obstructions in the compressor-inlet annulus. The presence of such obstructions at or near the survey plane can create local circumferential static-pressure gradients that introduce appreciable error, as indicated in reference 16. Details of the rake designs for total-temperature and total-pressure instruments are discussed in references 12 to 14 and 16.

For installations in which radial pressure gradients are ±3 percent or less and circumferential distributions are uniform, two to four total-pressure rakes with accompanying wall static-pressure taps enable the calculation of air flow within ±2 percent.

Any seal leakage or bleed flow from the engine must, of course, be measured or estimated and deducted from the inlet flow in order to determine gas flow in the tail pipe. One source of leakage that has been overlooked in some installations is the leakage out of the inlet duct downstream of the air-measuring station into the engine compartment. Some installations may actually provide for a small amount of leakage in order to cool the engine compartment, whereas others suffer an appreciable amount of leakage through holes in the compressor casing for lifting eyes or through slots in the inlet duct for such apparatus as retractable-screen actuating linkages. Although such apertures might appear to cause only small amounts of leakage, one production engine

investigated had as much as 5-percent air-flow leakage at high subsonic Mach numbers through openings in the retractable-inlet-screen housing and holes in the compressor casing. It is, of course, impossible to correctly determine engine net thrust if leakage of an unknown amount exists either upstream or downstream of the air-flow measuring station.

Equation for air flow at turbine nozzle. - The air flow may be determined from measurements at the choked turbine nozzle by the following equation:

$$m_1\left(1 + \frac{m_f}{m_1}\right) = C \frac{P_{th}A_{th}}{\sqrt{gRT_{th}}} \tag{6}$$

where C is a function of γ. Measurement of air flow at the choked turbine nozzle requires a knowledge of the product CA_{th} and measurements of turbine-inlet pressure and temperature (in lieu of pressure and temperature at the turbine throat). The product of C and effective turbine area can be determined only by calibration on an engine test stand or some other facility in which an independent and accurate measure of air flow is available. The most accurate method for obtaining turbine-inlet temperature is to calculate it from measured compressor-inlet, compressor-outlet, and turbine-outlet temperatures by the following equation:

$$\text{(Average turbine-inlet temperature)} = \text{(Turbine-outlet temperature)} +$$

$$\frac{c_{p,C}}{c_{p,T}}\left(\frac{1}{1 + \frac{m_f}{m_a}}\right)\left[\text{temperature rise across compressor} + \frac{m_{bl}}{m_a}\frac{c_{p,bl}}{c_{p,C}}\text{(temperature rise of bleed air)}\right]$$

where $c_{p,C}$ and $c_{p,T}$ are the specific heats averaged between the inlet and the outlet. The bleed flows and temperature rise may be estimated with sufficient accuracy from the manufacturer's manual.

Instrumentation for air-flow determination at turbine nozzle. - Instrumentation suitable for determining compressor-inlet temperature and pressure has been previously described in this section. Because large gradients in temperature exist at the turbine outlet, and also in some cases at the compressor outlet (when compressor-inlet pressure distribution is nonuniform), extensive surveys are also required at these stations for acceptable accuracy. The instruments required at the turbine outlet are discussed in detail in the section Turbojet with

Variable Exhaust Nozzle. At the compressor outlet, two or three rakes of two thermocouples equally spaced around the circumference are usually adequate for determining the average compressor-outlet temperature.

The fuel-air ratio m_f/m_a may be estimated with adequate accuracy from the manufacturer's performance manual or may be calculated from measured fuel flow and an assumed air flow. The fuel-air ratio so obtained may be considered a trial value and may be used in a first calculation of turbine-inlet temperature. Usually only one calculation of turbine-inlet temperature with the trial fuel-air ratio is required, but in some cases two or three iterations are necessary for adequate accuracy. Because the pressure gradients present are usually small, the turbine-inlet pressure may be measured with an integrating rake similar to that discussed in the section Determination of jet thrust from turbine-outlet pressure.

If the effective turbine area is known precisely, the air flow calculated from the choked-flow equations with the methods of temperature and pressure measurement described herein has been found in laboratory experiments to have an accuracy of ±2 to 3 percent.

There is little difference in the accuracy or complication of either of the methods of air-flow measurements. The choice of a method will therefore depend primarily upon the accessibility for instrument installation, the amount of inlet pressure gradient, and other factors peculiar to the airplane on which it is to be applied.

Summary of air-flow measuring methods. - Two techniques have been used for measuring air flow. For the first technique, instruments are installed at the engine inlet. The engine air flow is determined by the continuity equation in the following form:

$$m_1 = \frac{p_1 A_1}{\sqrt{gRT_1}\left(\frac{p_1}{P_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}}} \sqrt{\frac{2\gamma_1}{\gamma_1 - 1}\left[1 - \left(\frac{p_1}{P_1}\right)^{\frac{\gamma_1 - 1}{\gamma_1}}\right]} \tag{5}$$

The measurements required are given in the following table:

Quantity measured	Type of instrument	Number of instruments	Remarks
Compressor-inlet total pressure, P_1	Total-pressure rake	See Remarks	18 to 32 tubes if inlet pressure distribution uniform within ±3 percent. Greater number required for less uniform distribution.
Compressor-inlet static pressure, p_1	Wall static tap	4 to 8	Stream static survey also required if inlet pressure gradients greater than ±3 percent.
Compressor-inlet total temperature, T_1	Thermocouple rake	2 or 3 probes	

The air flow may be determined by this method with an accuracy of ±2 percent if the inlet pressure distribution is uniform within ±3 percent. Limited experience indicates that, in the presence of pressure gradients at the inlet of ±10 percent, the accuracy decreases unless additional instrumentation is supplied.

The second technique for measuring air flow involves the use of the choked turbine stators. The effective turbine-throat area must be determined by independent calibrations. With the effective turbine-throat area known, the air flow may be determined from the following equation:

$$m_1\left(1 + \frac{m_f}{m_1}\right) = C \frac{P_{th}A_{th}}{\sqrt{gRT_{th}}} \tag{6}$$

The measurements required are given in the following table:

Quantity measured	Type of instrument	Number of instruments	Remarks
Turbine-outlet total temperature, T_3	Thermocouple rake	4 rakes of 4 to 6 thermocouples	
Compressor-outlet total temperature	Thermocouple rake	2 or 3 rakes of 2 thermocouples	
Compressor-inlet total temperature, T_1	Thermocouple rake	Same as previous table	
Fuel-air ratio, m_f/m_a			Estimated from engine manufacturer's performance manual
Turbine-inlet total pressure	Integrating rake	1 rake	

Experience has shown that an accuracy of ±2 to 3 percent may be expected from the use of this method.

Turbojet with Fixed Exhaust Nozzle

Net-thrust equation. - The net thrust for the simple turbojet installation with a fixed exhaust-nozzle area and no afterburner or ejector (fig. 1(a)) may be determined by the following relation:

$$F_n = C_F \left[m_3V_3 + A_3(p_3 - p_0) \right] - m_1V_0 \tag{7}$$

Equation (7) is similar to equation (4) except for the coefficient C_F, which is hereinafter called the thrust coefficient. The thrust coefficient, which is discussed more fully in the section Determination of jet thrust from turbine-outlet pressure, is introduced to account for pressure losses between the instrumentation station and the nozzle exit, as well as differences between actual and measured velocities, areas, and pressures that inevitably result from the use of limited instrumentation. The measurements other than the standard aircraft measurements of ambient pressure and flight speed that are required in order

to determine the thrust from equation (7) are: (1) engine air flow, (2) pressure and temperature in the tail pipe, and (3) exhaust-nozzle area. Techniques for measuring each of these variables are discussed in detail.

Jet-thrust equation. - The tail-pipe measurements necessary to determine jet thrust (first term of eq. (7)) are now considered. Because of the possible errors in air-flow measurement just discussed, it is desirable to eliminate the air flow as one of the variables required for determining jet thrust. Similarly, because of the presence of radial and circumferential temperature gradients at the turbine outlet, it is also desirable to eliminate tail-pipe temperature as a necessary measurement to determine jet thrust. These two variables can be removed by rearranging the jet-thrust expression of equation (2) in the form

$$F_j = C_F\left[\gamma_3 A_3 p_3 M_3^2 + A_3(p_3 - p_0)\right] \tag{8}$$

For nearly all flight conditions at which thrust measurements are required, the flow in the exhaust nozzle is choked. The Mach number at the nozzle exit M_3 is therefore unity, and the static pressure at the nozzle throat can be expressed in terms of nozzle-outlet total pressure P_3 and the ratio of specific heats γ_3 as follows:

$$p_3 = P_3\left(\frac{2}{\gamma_3 + 1}\right)^{\frac{\gamma_3}{\gamma_3 - 1}} \tag{9}$$

Substitution into equation (8) gives

$$F_j = C_F A_3\left[(\gamma_3 + 1)\left(\frac{2}{\gamma_3 + 1}\right)^{\frac{\gamma_3}{\gamma_3 - 1}} P_3 - p_0\right] \tag{10}$$

The terms in equation (10) involving the specific heat γ may be replaced by a constant with reasonable accuracy (see refs. 17 to 19). For the nonafterburning engine with exhaust-gas temperatures ranging from about 1100° to 1700° R, the value of γ_3 varies from about 1.36 to 1.32, but this variation in γ_3 causes the coefficient

$(\gamma + 1)\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma}{\gamma - 1}}$ to vary only 0.4 percent. Replacing the specific-heat terms by the constant 1.25 results in only a small error over the entire range of exhaust-gas temperatures encountered. As will be

discussed in the section Determination of jet thrust from turbine-outlet pressure, the value of the specific-heat function of 1.25 may also be used for afterburning engines. An example from reference 18 of the correlation of jet thrust F_j with nozzle-pressure-drop parameter $(1.25P_3 - p_0)$ is presented in figure 3. Data such as these indicate that, after such a correlation is obtained, jet thrust can be determined by measuring only the exhaust-nozzle total pressure P_3 and the free-stream static pressure p_0. The jet thrust as expressed by equation (10) is affected only by the pressure-drop parameter $(1.25P_3 - p_0)$, the nozzle area, and the thrust coefficient. Therefore, after the calibration of figure 3 is obtained, the jet thrust will be unaffected by changing engines in the airplane provided radical changes in pressure distribution in the tail pipe do not result and provided the nozzle area is unchanged. If, upon changing engines, the nozzle area must be altered slightly, this area change can be compensated for by adjusting the jet-thrust values of the calibration curve in proportion to the area variation.

Instrumentation for jet-thrust determination. - The instrumentation recommended for measurement of the nozzle-inlet total pressure P_3 consists of two to four survey rakes of the type illustrated in figure 4. As an alternate method, a single swinging total-pressure probe located near the nozzle exit, as was utilized in reference 15, can be employed to measure the pressure at the nozzle exit. If rakes of the type indicated in figure 4 are used, each rake should include six to eight total-pressure tubes located on centers of equal annular areas. It is desirable to extend the survey as near the tail-pipe wall as possible. Such measurements can best be accomplished with a limited amount of instrumentation by radially offsetting the tubes in each rake from those in the others, in the same manner as was suggested for obtaining total-pressure measurements at the engine inlet. The axial location of the rake in the exhaust nozzle is not critical. In order to avoid large drag losses, the rakes are usually placed far enough upstream that the entire rake is surrounded by a low-velocity gas stream. This requirement generally results in placing the pressure-measuring tubes near the exhaust-nozzle inlet.

The free-stream static pressure used for computing jet thrust can be obtained from the standard airplane instrumentation. It is important that this pressure value not be influenced by local shock waves or other pressure fields.

If the nozzle-inlet total-pressure distribution is not altered appreciably by variations in flight conditions, a jet-thrust correlation with nozzle-pressure-drop parameter can be made, as suggested in reference 18, with only one or two total-pressure probes used. Such a

correlation would then allow removal of all but these pressure tubes from the tail pipe for flights subsequent to determination of the correlation.

If the exhaust-nozzle-exit area A_3 is measured while the nozzle is cold, a correction must be made for thermal expansion of the nozzle during engine operation; nozzle skin temperatures as high as 1000° to 1200° F may be obtained at some operating conditions. Because the skin temperature is affected by engine operating condition and external circulation in the engine compartment, it is recommended that skin temperature of the nozzle be measured. The increase in area above the cold measurement can then be obtained in the conventional manner by applying the proper coefficient of expansion to the measured skin-temperature difference.

The thrust coefficient as used herein relates the actual thrust to the thrust determined from pressure measurements in the exhaust nozzle. Experience has shown that, if a reasonably complete pressure survey is made in the exhaust nozzle in the manner suggested in the previous discussion, the value of thrust coefficient for simple conical nozzles is constant for exhaust-nozzle pressure ratios greater than 1.9 and a value of about 0.98 can be used with a probable error of about ±1 percent. Of course, this value of thrust coefficient will not be valid if an exhaust nozzle is used that permits discharge of exhaust gases in directions other than the thrust direction, such as might be the case for a nozzle with a nonplanar discharge, or if a nozzle is used that has large leakage or other losses between the pressure measuring station and the outlet. The thrust coefficient in these cases must be determined by calibration of the individual nozzles on a thrust stand.

With the exception of the specific-heat ratio, the jet thrust is directly proportional, or nearly so, to each variable in equation (10). Therefore, if an accuracy of ±1 percent in jet thrust is desired, each variable must be measured with a probable error of no more than ±1 percent. There should be little difficulty in accurately determining the exhaust-nozzle-exit area A_3 or the free-stream static pressure p_0; and, as previously mentioned, the jet thrust is relatively insensitive to substantial variations in specific-heat ratio γ. If there is essentially no error in these measurements, the accuracy of the jet thrust becomes primarily a function of the accuracy with which the nozzle-inlet total pressure P_3 can be measured. The average nozzle-inlet total pressure used in equation (10) is an area-weighted average of the total pressures measured at this station. A typical example of nozzle-inlet total-pressure gradients encountered in current engines is presented in figure 5. Variations in radial pressure gradient encountered range from about 1 to 40 percent, varying with engine type and operating condition. The reduction in total pressure near the center of the jet, which is

generally significant on most engines, results from separation of the flow from the tail-pipe centerbody. These data indicate that a rather complete total-pressure survey of the type recommended is necessary in order to obtain an accurate measurement of total pressure in the tail pipe.

Summary of net-thrust instrumentation for fixed exhaust nozzle. - Combining the simplified expression for jet thrust with equation (4) for net thrust gives the following equation for net thrust for a turbojet engine with a fixed exhaust nozzle:

$$F_n = C_F A_3 (1.25P_3 - p_0) - m_1 V_0$$

The required measurements other than air flow, nozzle area, and ambient conditions are given in the following table:

Quantity measured	Type of instrument	Number of instruments	Remarks
Exhaust-nozzle-outlet total pressure, P_3	Total-pressure rake	2 to 4 rakes of 6 to 8 probes in each rake	After proper calibration, rakes may be replaced by one or two probes.
Thrust coefficient, C_F	Experimentally determined		A value of 0.98 may be used for most conical exhaust nozzles.

An accuracy of ±2 to 3 percent of net thrust is probably obtained by this method.

Turbojet with Variable Exhaust Nozzle

Net-thrust equation. - Equation (7) may be used to determine net thrust for a turbojet engine with either a fixed or variable-area exhaust nozzle. As previously discussed, in order to use equation (7) the exhaust-nozzle area must be known. In some cases for an engine with a variable-area exhaust nozzle, it may be more convenient or accurate to determine thrust from pressure and temperature measurements in

the exhaust nozzle rather than from measurements of pressure and exhaust-nozzle area. In order to use the temperature to determine net thrust, equation (7) is modified as follows:

$$F_n = C_V m_3 \sqrt{2g \frac{\gamma_3}{\gamma_3 - 1} R_3 T_3 \left[1 - \left(\frac{p_0}{P_3}\right)^{\frac{\gamma_3 - 1}{\gamma_3}}\right]} - m_1 V_0 \qquad (11)$$

The term under the radical is the jet velocity for isentropic expansion and may be used without appreciable error for installations with a convergent exhaust nozzle for pressure ratios P_3/p_0 up to about 4. For higher pressure ratios, if a convergent nozzle is used, an effective jet velocity from reference 19 that takes into account the departure from complete isentropic expansion may be used in place of the radical. The velocity coefficient C_V is discussed in the following section.

Instrumentation for jet-thrust determination. - Although the exhaust-nozzle area is not required in order to compute the jet thrust by equation (11), the exhaust-gas temperature must be measured. Thermocouples should therefore be installed on the total-pressure survey rakes located near the exhaust-nozzle exit. It is recommended that four to six thermocouples be installed on each of three or four rakes so that both radial and circumferential temperature variations in the average measured value may be considered. If the thermocouples are installed on the centers of equal annular areas, the measured temperatures can be arithmetically averaged, but, if they are not, an area-weighted average temperature should be computed.

In addition to measuring the principle variables thus far discussed, values of exhaust-nozzle velocity coefficient and ratio of specific heats for the exhaust gases must also be determined. Numerous determinations of velocity coefficient have been made for variable-area nozzles at various simulated flight conditions. It has been found that the velocity coefficient is dependent on the type of instrumentation used, the nozzle design, and the nozzle pressure ratio. Velocity coefficients for typical variable-area exhaust nozzles are given in reference 1. The velocity coefficient for a current well-designed variable-area nozzle (planar exhaust area) is constant at a value of about 0.975 for nozzle pressure ratios greater than 2.0. The same value has been found to hold for simple iris-type nozzles. This value of velocity coefficient may be used with a probable error of about ±1 percent for variable-area exhaust nozzles similar to those discussed in reference 1 if the instrumentation described herein is used. For

nozzles radically different from the simple iris or the planar variable-area nozzle discussed in reference 1, the value of velocity coefficient may be determined only by calibration.

Jet-thrust measurements with a variable-area exhaust nozzle are generally less accurate than those with a fixed nozzle, because two engine variables (temperature and pressure at the exhaust-nozzle outlet) must be measured in order to compute the variable-nozzle thrust (eq. (11)) as compared with the single variable of nozzle-inlet total pressure that must be measured for the fixed-nozzle installation. If the influence of these variables on the variable-nozzle thrust measurement is considered, examination of equation (11) shows that jet thrust varies proportionally with gas flow, nearly proportionally with nozzle inlet pressure, and as the square root of exhaust-gas temperature. Experience has indicated that the greatest accuracy with which each of these variables can be measured is generally on the order of ±1 percent. Thus, if there are no compensating errors, the accuracy of the jet-thrust measurement would seldom be better than about $\pm 1\frac{1}{2}$ percent.

Summary of net-thrust instrumentation for variable-area exhaust nozzle. - Net thrust may be determined by the following equation:

$$F_n = C_V m_3 \sqrt{2g \frac{\gamma_3}{\gamma_3 - 1} R_3 T_3 \left[1 - \left(\frac{p_0}{P_3}\right)^{\frac{\gamma_3 - 1}{\gamma_3}}\right]} - m_1 V_0 \tag{11}$$

The measurements required are listed in the following table:

Quantity measured	Type of instrument	Number of instruments	Remarks
Exhaust-nozzle-outlet total temperature, T_3	Thermocouple rake	3 or 4 rakes of 4 to 6 thermocouples	
Exhaust-nozzle-outlet total pressure, P_3	Total-pressure rake	2 to 4 rakes of 6 to 8 probes each	
Velocity coefficient, C_V			Approx. 0.975 for variable nozzles of conventional designs. For unconventional designs a calibration is required.

The accuracy to be expected is ±2 to 3 percent in net thrust.

Turbojet with Variable Exhaust Nozzle and Afterburner

Net-thrust equation. - With an afterburning turbojet engine, direct measurement of the exhaust-gas temperature is impractical. It is therefore necessary to determine the exhaust-nozzle area and apply equation (7) in order to determine the net thrust

$$F_n = C_F\left[m_3V_3 + A_3(p_3 - p_0)\right] - m_1V_0$$

This equation is the same as equation (4) except that the secondary flow is assumed to be zero. The same variables that were required for the fixed-nozzle installation must therefore be measured, but the introduction of the afterburner further complicates the measurement of the exhaust-nozzle-inlet total pressure and the exhaust-nozzle area. Air flow is determined by the methods previously described.

Instrumentation for net-thrust determination. - The exhaust-nozzle area used for the net-thrust determination is the projected axial-discharge area corrected for thermal expansion. Temperatures measured by thermocouples attached to the nozzle can be used to determine the thermal-expansion correction. A calibration of projected nozzle-exit area with nozzle position is also required. The projected area may be determined with sufficient accuracy by direct measurements or by computation of the area from scale drawings of the nozzle.

In order to determine nozzle-inlet total pressure, either a direct measurement can be made in flight, or the nozzle-inlet pressure may be determined from a calibration with turbine-outlet total pressure, as will be discussed in more detail in the following section. For a direct measurement, a single survey rake across the diameter of the nozzle such as the one illustrated in figure 6 may be used. Generally, at least 10 total-pressure tubes (see refs. 14 and 16 for detailed pressure-rake design rules) are required for accurate measurements from such a survey across the nozzle. Because the rake is located in the extremely hot gas stream, some method of cooling is necessary. In laboratory setups water cooling is generally employed; for flight use a water-cooled rake may not be practical. The nozzle-inlet total pressure can also be measured with a swinging probe, as suggested in reference 15. Such a probe need not be cooled for adequate life if it is immersed in the hot gas stream for less than about 4 seconds per traverse.

Determination of jet thrust from turbine-outlet pressure. - In order to eliminate the need for a cooled exhaust-nozzle survey rake or a swinging probe in flight, the exhaust-nozzle-outlet total pressure may

be obtained from a calibration. The exhaust-nozzle-outlet total pressure may be calibrated against a total pressure measured upstream of the afterburner fuel injectors, where the gas temperatures are low. The jet thrust may then be determined by the following equation, which is similar to equation (10):

$$F_j = C_F A_3 \left[1.25 P_2 \left(1 - \frac{P_2 - P_3}{P_2}\right) - p_0\right] \tag{12}$$

The loss ratio $(P_2 - P_3)/P_2$ and the turbine-outlet pressure P_2 must therefore be determined. An absolute average value of the turbine-outlet pressure is not necessary for this calibration, but only a reference total pressure P_2 that maintains a consistent relation to nozzle-inlet pressure for all flight conditions. Any differences between absolute average pressure and the reference pressure are accounted for in the calibration. A single integrating total-pressure probe of the type illustrated in figure 7 may thus be employed except in the rare cases where variations in flight conditions appreciably affect turbine-discharge pressure distribution.

The loss ratio $(P_2 - P_3)/P_2$ may be determined by calibration of turbine-outlet pressure P_2 with nozzle-inlet pressure P_3 at limiting turbine-outlet temperature and with choked flow in the exhaust nozzle. With these conditions satisfied, the total-pressure-loss ratio from the turbine outlet to the exhaust-nozzle inlet is essentially a function of afterburner-inlet Mach number (which varies only slightly with flight condition) and afterburner temperature ratio (which varies in proportion to the square of exhaust-nozzle-exit area). As a result, the afterburner total-pressure-loss ratio can be correlated as an approximately linear function of exhaust-nozzle-exit area. A typical correlation of this type is presented in figure 8, which shows the variation of afterburner total-pressure-loss ratio with exhaust-nozzle-exit area ratio for several flight conditions. There is little effect of flight condition on the correlation; the slightly higher pressure losses at high altitudes are undoubtedly associated with a slight increase in afterburner-inlet Mach number with altitude, which is typical of most engines.

In order to determine the jet thrust by equation (12) at any flight condition, a pressure-loss calibration similar to that of figure 8 and a calibration of jet thrust with pressure-drop parameter $(1.25P_3 - p_0)$ as shown in figure 9 must be established. In figure 9, the jet thrust per unit of projected nozzle area is correlated with the pressure-drop parameter as shown by the typical afterburner data for several simulated flight conditions. In this case, as is typical, the data fall along a single line, indicating that the thrust coefficient C_F, as defined

herein, is constant for all nozzle areas and is independent of flight conditions and of afterburner operation. Because this correlation exists, it is evident that a similar calibration could be made in an airplane during ground tests with more extensive instrumentation than could be used in flight. A relatively simple thrustmeter, such as the type suggested in reference 18, could then be utilized as a cockpit instrument to indicate jet thrust during flight.

The thrust coefficient C_F for two nozzles, a clamshell nozzle (reported in ref. 1) and an iris nozzle, are presented in figure 10. These thrust coefficients were determined by using a pressure rake similar to that of figure 6 to measure P_3. Photographs of the nozzles are shown in figure 11. There is considerable scatter in the data of figure 10, but the average thrust coefficient of the clamshell nozzle was within ±1 percent of 0.975 for nozzle pressure ratios of 2.0 to 4.0. Also, the thrust coefficient of the iris nozzle was about 0.97 over the same range of pressure ratios. The thrust coefficient, as defined herein, is essentially insensitive to variations in nozzle pressure ratio; therefore, these values are also applicable for higher pressure-ratio conditions.

All variable-area nozzles will not have thrust coefficients equal to those shown, but well-designed nozzles similar to the ones investigated will have thrust coefficients equivalent to those values if the appropriate pressures are determined by the instrumentation recommended herein. If a nozzle is somewhat different or unusual in design or if widely different instrumentation is used, the only method of determining the thrust coefficient is by calibrating the nozzle and instrumentation on a thrust stand.

The thrust determined by equation (12) is not sensitive to the value of γ used. A value of γ resulting in a multiplying factor of 1.25 (eq. (12)) may be used over the entire range of afterburner-outlet temperatures with a maximum error of ±1 percent and with an error of less than ±0.5 percent over most of the temperature range. If a more precise value of specific-heat ratio is desired, the exhaust-gas temperature must be calculated. For this calculation a mean value of 1.25 may be assumed for the specific-heat ratio. With choked flow in the exhaust nozzle, the temperature may be obtained from the equation

$$T_g = \frac{P_3^2 A_3^2 \gamma_3}{g m_3^2 R_3} \left(\frac{2}{\gamma_3 + 1}\right)^{\frac{\gamma_3+1}{\gamma_3-1}}$$

where the gas flow can be taken as the sum of the inlet air flow and the afterburner fuel flow. In this case, the primary-engine fuel flow is assumed to be approximately equal to the unmeasured air leakage from the engine. From this temperature and the information in reference 19, a more precise value of specific-heat ratio can be selected.

This procedure will probably provide an accuracy of about ±1 percent in the absolute value of nozzle-inlet total temperature in most installations. Such an error in temperature would result in an error in jet thrust varying from slightly less than ±1 percent at a nozzle pressure ratio of 1.9 to about ±0.5 percent at a pressure ratio of 10. Additional errors are introduced, of course, by errors in area and air-flow measurements. Experience indicates that an over-all accuracy of about ±2 to 3 percent is common in afterburner installations.

<u>Summary of net-thrust instrumentation for afterburning engine</u>. - The net thrust with an afterburning engine may be determined by the following equation:

$$F_n = C_F A_3 \left[1.25 P_2 \left(1 - \frac{P_2 - P_3}{P_2}\right) - p_0\right] - m_1 V_0$$

The measurements required are as follows:

Quantity measured	Type of instrument	Number of instruments	Remarks
Pressure-loss ratio, $(P_2 - P_3)/P_2$	From calibration		
Turbine-outlet total pressure, P_2	Integrating rake	1 rake	
Thrust coefficient, C_F			Approx. 0.97 for variable nozzles of conventional design. For unconventional designs a calibration is required.

The accuracy of the net thrust to be expected by the use of this method is ±2 to 3 percent.

Afterburning and Nonafterburning Turbojet with Variable Exhaust Nozzle and Ejector

The methods of measuring net thrust of a turbojet engine with and without afterburning but without an ejector have been discussed. Addition of an ejector to the engine installation further complicates the task of measuring the installation net thrust, because both the inlet momentum drag and the jet thrust of the secondary or cooling air passing through the ejector must be considered. The jet thrust of the secondary air, with expansion to free-stream pressure assumed, cannot be added directly to the primary-nozzle jet thrust because interaction of the two streams with possible overexpansion of the primary jet within the ejector can have large effects on the over-all jet thrust. It is necessary, therefore, either to integrate the pressure and temperature at the ejector exit or to calculate the contributions of the primary and secondary stream in such a manner as to account for overexpansion or interaction effects. Because of the difficulties and complications involved in accurately integrating the pressures and temperatures at the ejector outlet (ref. 15), an alternate procedure for computing the contributions of the primary and secondary streams, accounting for overexpansion and interaction effects, is considered herein.

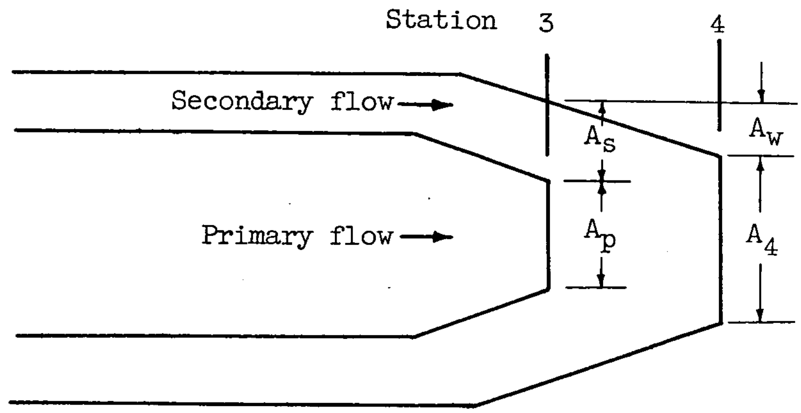
Net-thrust equation. - The net thrust of the ejector installation can be expressed as

$$F_n = F_j - m_1 V_0 - m_s V_0$$

As mentioned previously, the free-stream velocity used for the secondary air may be equal to the actual free-stream velocity. However, if the secondary air is removed from the boundary layer along the fuselage, a large amount of the free-stream momentum of the secondary air might be chargeable to airplane drag and the velocity used would thus be the average velocity approaching the secondary inlet. If the latter case is considered, a total-pressure survey and a static-pressure measurement are required just upstream of the secondary inlet.

The amount of instrumentation required in order to obtain an accurate velocity depends, of course, on the type of inlet and the condition of the approaching flow. In addition, a total-pressure survey and static-pressure and -temperature measurements are also required at or downstream of the secondary inlet in order to measure the secondary air flow. The location and type of instrumentation should be similar to that previously described for measuring engine air flow but will depend on the installation geometry. Because the inlet momentum of the secondary air is generally a very small portion of the net thrust, a high degree of accuracy is not required in measuring these values.

Jet-thrust equation. - In order to clarify the development of the jet-thrust equation for an ejector installation, the following sketch is included:



(c) Typical ejector configuration.

For this configuration station 3 is defined as the plane of the primary-nozzle exit, station 4 is the ejector exit, and the area A_w represents the projected area of the ejector shroud between stations 3 and 4. The jet thrust of the ejector is expressed by the one-dimensional equation

$$F_j = m_4V_4 + A_4(p_4 - p_0) \tag{13}$$

Balancing the momentum and pressure forces acting between stations 3 and 4 then gives the following equation for jet thrust:

$$F_j = C_F\left[m_{3,p}V_{3,p} + A_p(p_{3,p} - p_0)\right] + m_sV_{3,s} + A_s(p_{3,s} - p_0) - \int_4^3 p_w\, dA_w + A_wp_0 \tag{14}$$

The first term of this equation comprises the primary-nozzle jet thrust, which can be determined as previously discussed, with the presence of the ejector neglected and the same thrust coefficient used. The remaining terms represent the thrust or drag contribution of the ejector and account for any overexpansion of the primary jet within the shroud.

The ejector thrust may be calculated by either of two slightly different equations, the choice depending upon the installation and the type of instrumentation most convenient to install. The jet thrust of

the primary nozzle in equation (14) is replaced by $F_{j,p}$ and a substitution is made for $m_s V_{3,s}$, resulting in the following expression for the ejector thrust:

$$F_j = F_{j,p} + \frac{m_s^2 g R_s t_{3,s}}{p_{3,s} A_s} + A_s(p_{3,s} - p_0) - \int_4^3 p_w \, dA_w + A_w p_0 \qquad (15)$$

In this equation, the measurements required, in addition to those of the previous installations, are:

(1) Secondary mass flow (also required for momentum drag as previously discussed)

(2) Static pressure in plane of primary-nozzle exit

(3) Static temperature of secondary flow near primary-nozzle exit

(4) Local wall static pressures within ejector between stations 3 and 4 (not required on cylindrical ejector, where projected area between stations 3 and 4 equals zero).

Inaccuracies are to be expected in measuring the secondary mass flow. In addition, accurate measurement of the secondary air temperature is often difficult to obtain because of radiation from the hot tail pipe to the thermocouples. In order to eliminate these two variables, the jet-thrust equation can be rearranged in the following manner:

$$F_j = F_{j,p} + p_{3,s} A_s \left(\frac{2\gamma_s}{\gamma_s - 1}\right)\left[\left(\frac{P_{3,s}}{p_{3,s}}\right)^{\frac{\gamma_s - 1}{\gamma_s}} - 1\right] + A_s(p_{3,s} - p_0) - \int_4^3 p_w \, dA_w + A_w p_0 \qquad (16)$$

Although the secondary temperature and secondary mass flow are both eliminated from this expression for jet thrust, they are replaced by the secondary total pressure and the ratio of specific heats of the secondary flow.

Instrumentation for determining jet thrust. - The momentum of the secondary flow at station 3, represented by the second term of equations (15) and (16), seldom exceeds about 10 percent of the jet thrust at subsonic flight speeds. Therefore, at these flight conditions a great deal

of accuracy is not required in measuring the secondary mass flow and secondary temperature when equation (15) is used, or the secondary total pressure and the ratio of specific heats of the secondary flow when equation (16) is used. The momentum of the secondary flow can exceed 10 percent of the jet thrust at supersonic Mach numbers when the secondary pressure ratio exceeds about 2.0. Sufficient accuracy in the secondary-temperature measurement can probably be obtained by using about six shielded thermocouple probes equally spaced around the secondary annulus and located at the primary-nozzle exit or a short distance upstream. Installation of four to six total-pressure probes located similarly in the secondary passage should give a sufficiently accurate total pressure. A specific-heat ratio of 1.4 can be used for the secondary flow.

The static pressures at station 3 and along the ejector shroud between stations 3 and 4 should be measured with a detailed survey of wall static-pressure orifices for convergent ejectors, because these pressures can significantly affect the jet thrust. Furthermore, at some operating conditions the axial static-pressure distribution in the ejector becomes very erratic because of overexpansion of the primary jet with a resultant shock system within the ejector. Such pressure distributions are presented and discussed in reference 9. At least four static wall orifices should be located around the shroud at station 3, and a number of wall orifices should be located along the ejector shroud. It is recommended that the orifices be located on equal axial spacings between stations 3 and 4, no more than approximately 1 to 2 inches apart; if there is sufficient space, this longitudinal survey should be duplicated in two circumferential locations.

If the secondary flow passage is obstructed by actuators or other impediments to the flow, accurate measurement of the secondary total and static pressures becomes very difficult and in some cases impossible. Therefore, this passage should be kept relatively free of such obstructions.

Because of the marked differences in ejector flow characteristics and pressure losses from one ejector configuration to another, experiments have been conducted to determine whether the jet thrust of an ejector can be determined accurately by the method discussed herein. A typical comparison of calculated values of jet thrust with values obtained from force measurements is presented in figure 12 for scale models of cylindrical, convergent, and divergent ejectors. These results are typical of the scale-model-ejector data presented in reference 9. The irregularities in the curve for a secondary pressure ratio of 1.0 in figure 12(a) are due to the presence of local shocks within the ejector. These data indicate the necessity for a large number of pressure tubes for accurate thrust determination. The calculated thrusts agree reasonably well with the measured values. The

agreement between the two values is generally within about 2 percent. It should be noted that there were no obstructions in the secondary flow passage and that a minimum amount of the recommended instrumentation was used.

Summary of net-thrust instrumentation for ejector installation. - Net thrust may be determined from the following equation:

$$F_n = F_{j,p} + p_{3,s}A_s \frac{2\gamma_s}{\gamma_s - 1}\left[\left(\frac{P_s}{p_{3,s}}\right)^{\frac{\gamma_s-1}{\gamma_s}} - 1\right] + A_s(p_{3,s} - p_0) - \int_4^3 p_w \, dA_w + p_0 A_w - m_1 V_0 - m_s V_s$$

The measurements required in addition to those for installation without ejectors are listed in the following table:

Quantity measured	Type of instrument	Number of instruments	Remarks
Secondary total pressure, P_s	Total-pressure probe	4 to 6 probes	
Secondary static pressure, $p_{3,s}$	Secondary-passage wall static	4 wall statics	
Secondary total temperature, T_s	Secondary-air temperature thermocouple	6 shielded thermocouples	
Wall statics, p_w	Wall static orifice	2 axial rows of orifices spaced 1 to 2 inches apart	None required for cylindrical ejector.

ENGINE-PERFORMANCE CALIBRATION

The discussion thus far has dealt with the thrust of the complete turbojet installation. As mentioned in the INTRODUCTION, if discrepancies are found between the thrust of the installed engine and the thrust anticipated from the design calculations, the contributions of the various parts of the installation must be isolated in order to determine the

source of the discrepancies. In order to isolate the parts of the system, the contribution of the engine itself must be determined. In this section a method of determining the isolated engine performance is discussed.

Pumping Characteristics

Because the function of the basic turbojet engine is to pump a quantity of air to a total pressure and temperature higher than the inlet total pressure and temperature, the performance of the engine can be described by what are commonly referred to as the engine pumping characteristics. This method of presenting engine performance was discussed in reference 20 and is illustrated in figure 13, which presents the variation of engine total-pressure ratio P_2/P_1 with engine total-temperature ratio T_2/T_1 and of the ratio of corrected air flow $W_a\sqrt{\theta_1}/\delta_1$ to the rated corrected air flow with the ratio of corrected engine speed $N/\sqrt{\theta_1}$ to rated corrected engine speed for a typical turbojet engine.

The performance characteristics of the installed engine expressed in the form of figure 13 can be compared directly with those of the engine manufacturer, or the parameters can be used in the conventional thrust equation to enable comparisons on a thrust basis.

Instrumentation for Determining Pumping Characteristics

The measurements of engine air flow and inlet total pressure and total temperature previously discussed can be used directly in the pumping characteristics. The exhaust-nozzle-inlet total-temperature measurements described for nonafterburning engines can be used in the engine-temperature-ratio parameter. For afterburning engines a temperature survey must be obtained between the afterburner fuel injectors and the turbine. Measurements of engine speed and turbine-outlet total pressure are also required.

The measurements of engine speed and turbine-outlet total pressure can be obtained with a relatively small amount of instrumentation. Engine speed should be measured with an accurate tachometer or a chronatach. Although there is often a considerable radial turbine-outlet total-pressure gradient, as illustrated by the data in figure 14 for three current engines, it has been found that an integrating rake such as the one shown in figure 7 provides turbine-outlet total-pressure measurements of sufficient accuracy if the flow swirl angles are less than approximately $\pm 10^{\circ}$ and if radial pressure gradients are less than approximately 25 percent. Therefore, installation of two or three such

integrating rakes near the turbine outlet will, in most cases, provide adequately accurate turbine-outlet total-pressure measurements.

In order to obtain an accurate average value of turbine-outlet temperature for the afterburning engine, a considerable number of thermocouples are required, because of the substantial radial and circumferential turbine-outlet temperature gradient. An example of the turbine-outlet temperature gradients for three current engines is presented in figure 15. It has been found that 15 to 30 thermocouples divided among at least three radial survey rakes are necessary to obtain accurate average temperature values.

As an alternate method of measuring turbine-outlet temperature, the temperature can be computed from engine fuel-flow measurements. This technique requires accurate measurements of fuel flow and a knowledge of the engine combustion efficiency. Thus, the enthalpy per pound of air at the turbine outlet can be expressed as

$$H_2 = H_1 + \eta_c h_f\left(\frac{m_f}{m_a}\right) \tag{17}$$

The turbine-outlet total temperature can be determined from conventional enthalpy charts. Equation (17) is true only when there is no significant quantity of air bled from the engine between the engine inlet and the turbine outlet.

In order to determine turbine-outlet temperature for the case of compressor-outlet bleed, equation (17) must be modified as follows:

$$H_2 = H_1 + \eta_c h_f\left(\frac{W_f}{W_a}\right) - H_{bl}\left(\frac{W_{bl}}{W_a}\right) \tag{18}$$

For this case, measurements of the flow rate and the bleed-air temperature are also required.

Reynolds Number Effects

Because of altitude or Reynolds number effects, performance data for all flight conditions will not generalize to a single set of performance curves such as the ones in figure 13. Generalization of the performance has been obtained, however, for given values of Reynolds number index $\frac{\delta_1}{\varphi_1\sqrt{\theta_1}}$. This parameter is a measure of the Reynolds number at the compressor inlet and is defined as the ratio of the Reynolds

number of the compressor-inlet flow at altitude to the Reynolds number of the compressor-inlet flow at standard sea-level conditions. The derivation and use of this parameter are discussed in reference 20.

CONCLUDING REMARKS

Techniques suitable for the experimental determination of thrust in prototype turbojet-powered airplanes have been discussed. Several types of power-plant installations have been included. Summary tables showing the type and number of instruments required for accurate thrust measurement are included after the discussion of each type of power-plant installation.

As an installation becomes more complicated by the addition of an afterburner, a variable-area exhaust nozzle, or an ejector, the required amount of instrumentation is increased and the methods of computing performance become more complex. If instrumentation techniques conforming to those recommended herein are utilized, it should be possible to measure the installation performance in flight with a degree of accuracy adequate for most purposes. There are no substitutes or short cuts that will eliminate extensive instrumentation for the determination of the thrust of an installed engine.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, April 14, 1955

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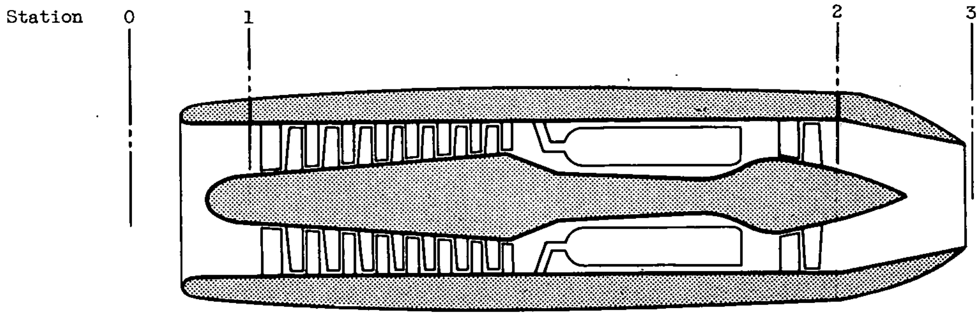
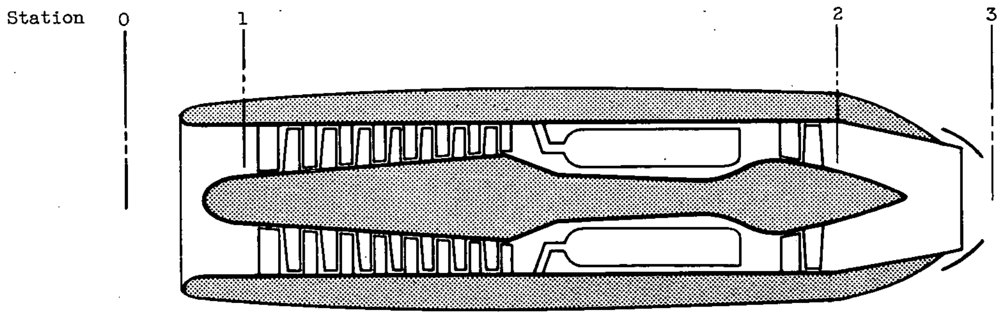
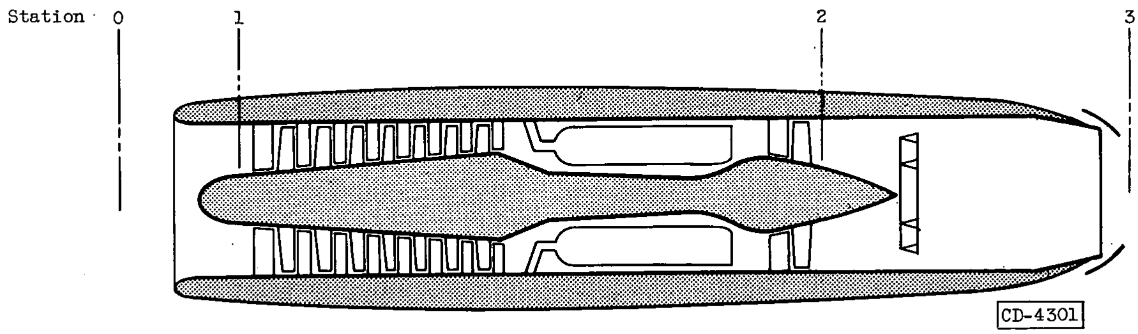


(a) Fixed exhaust nozzle, no afterburner, no ejector.

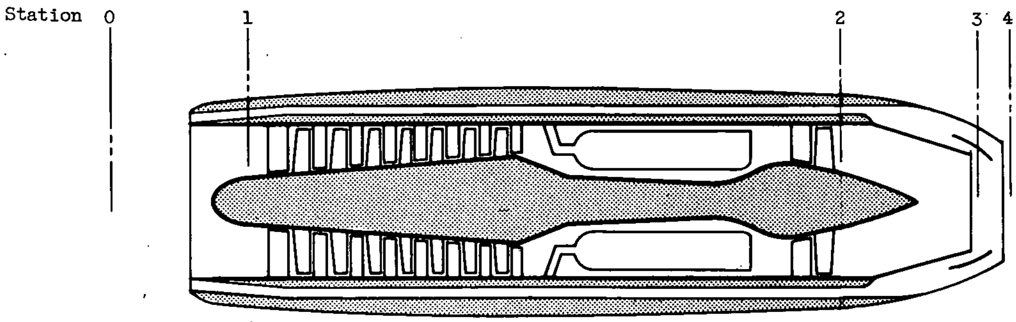


(b) Variable exhaust nozzle, no afterburner, no ejector.

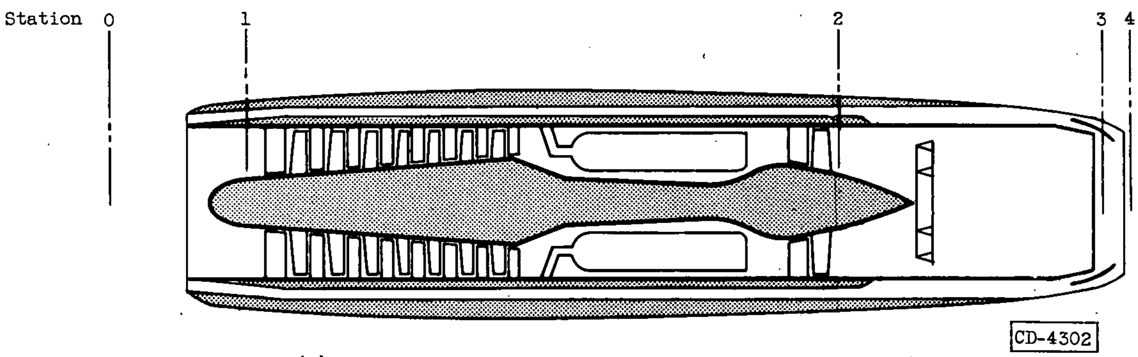


(c) Variable exhaust nozzle, with afterburner, no ejector.

Figure 1. - Schematic drawings of turbojet installations.



(d) Variable exhaust nozzle, no afterburner, with ejector.



(e) Variable exhaust nozzle, with afterburner, with ejector.

Figure 1. - Concluded. Schematic drawings of turbojet installations.

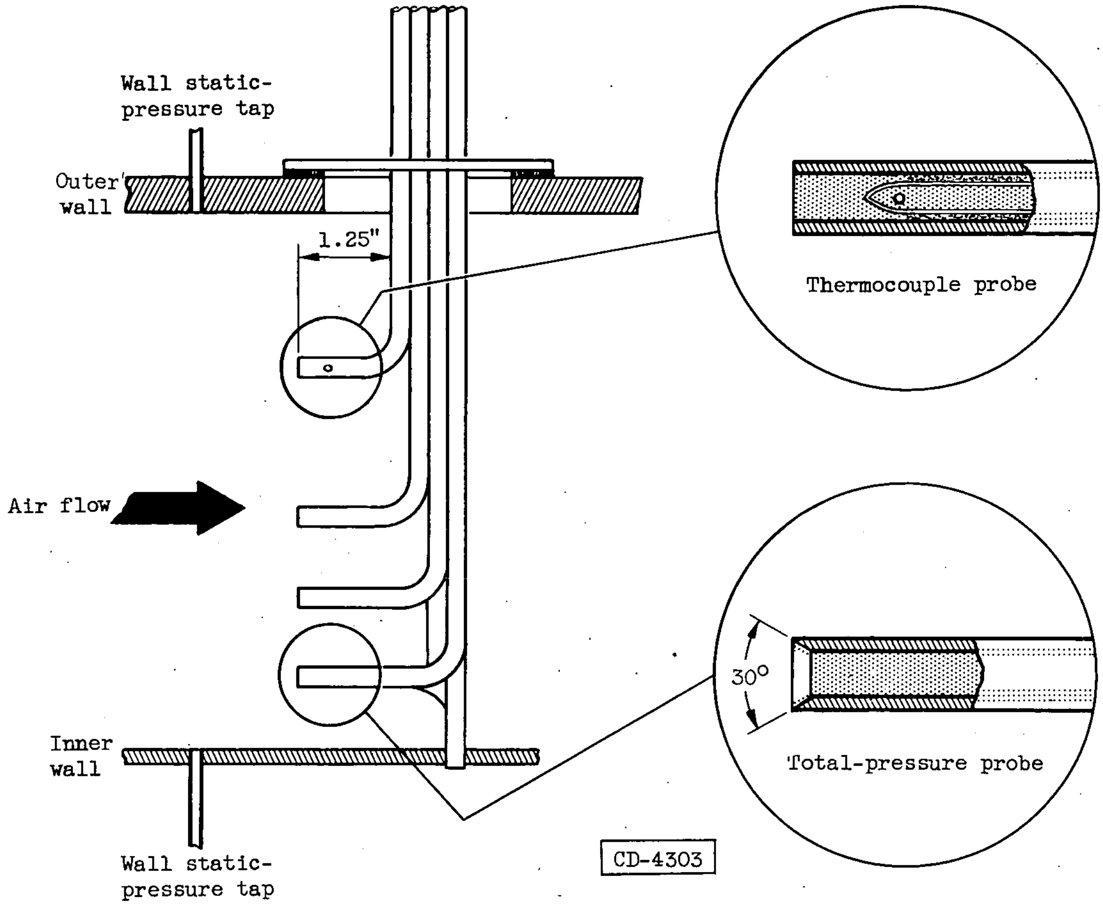


Figure 2. - Schematic drawing of compressor-inlet instrumentation. (Radial spacing of total-pressure probes based on equal areas.)

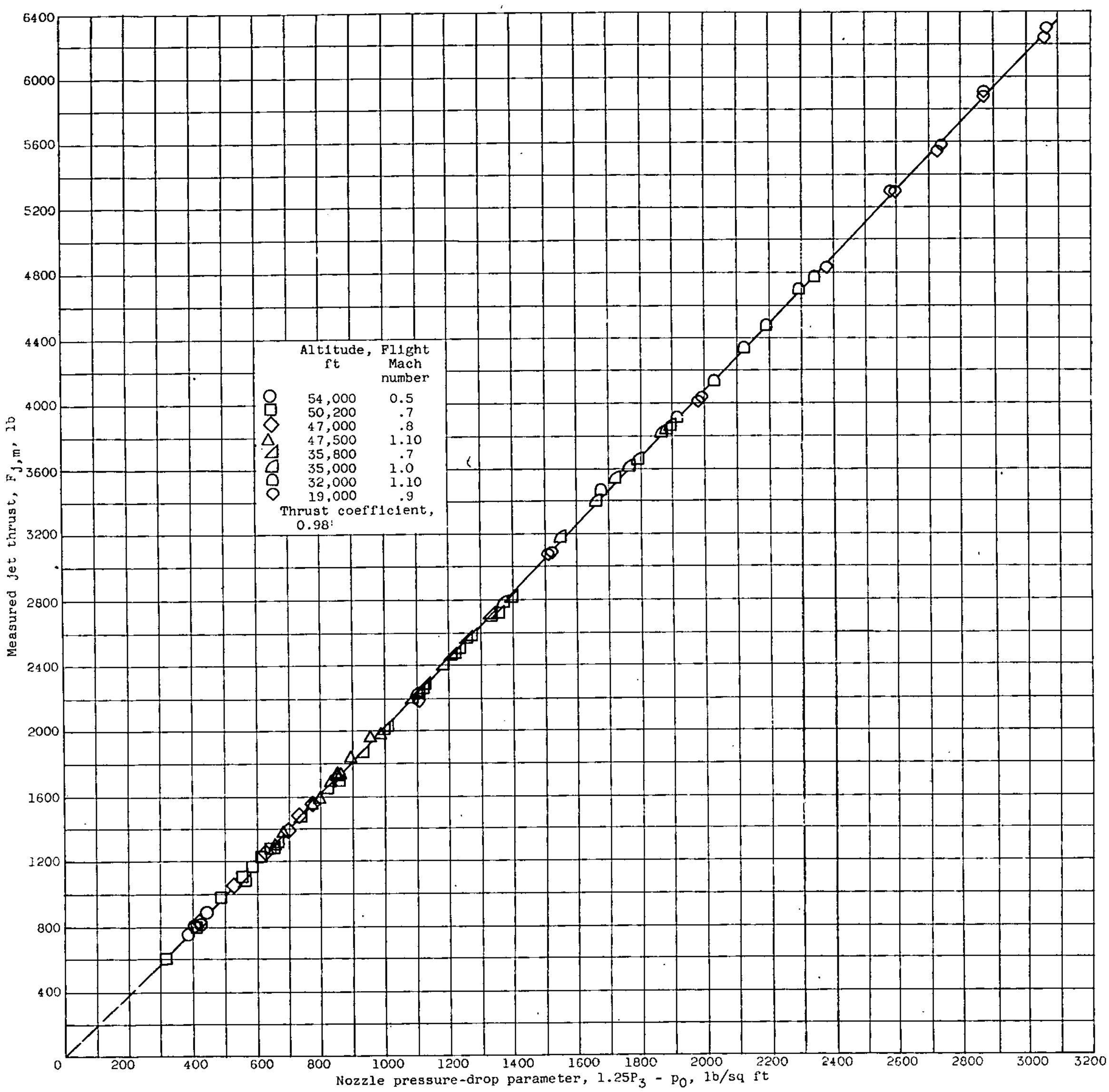


Figure 3. - Jet-thrust correlation for engine having fixed-area exhaust nozzle.

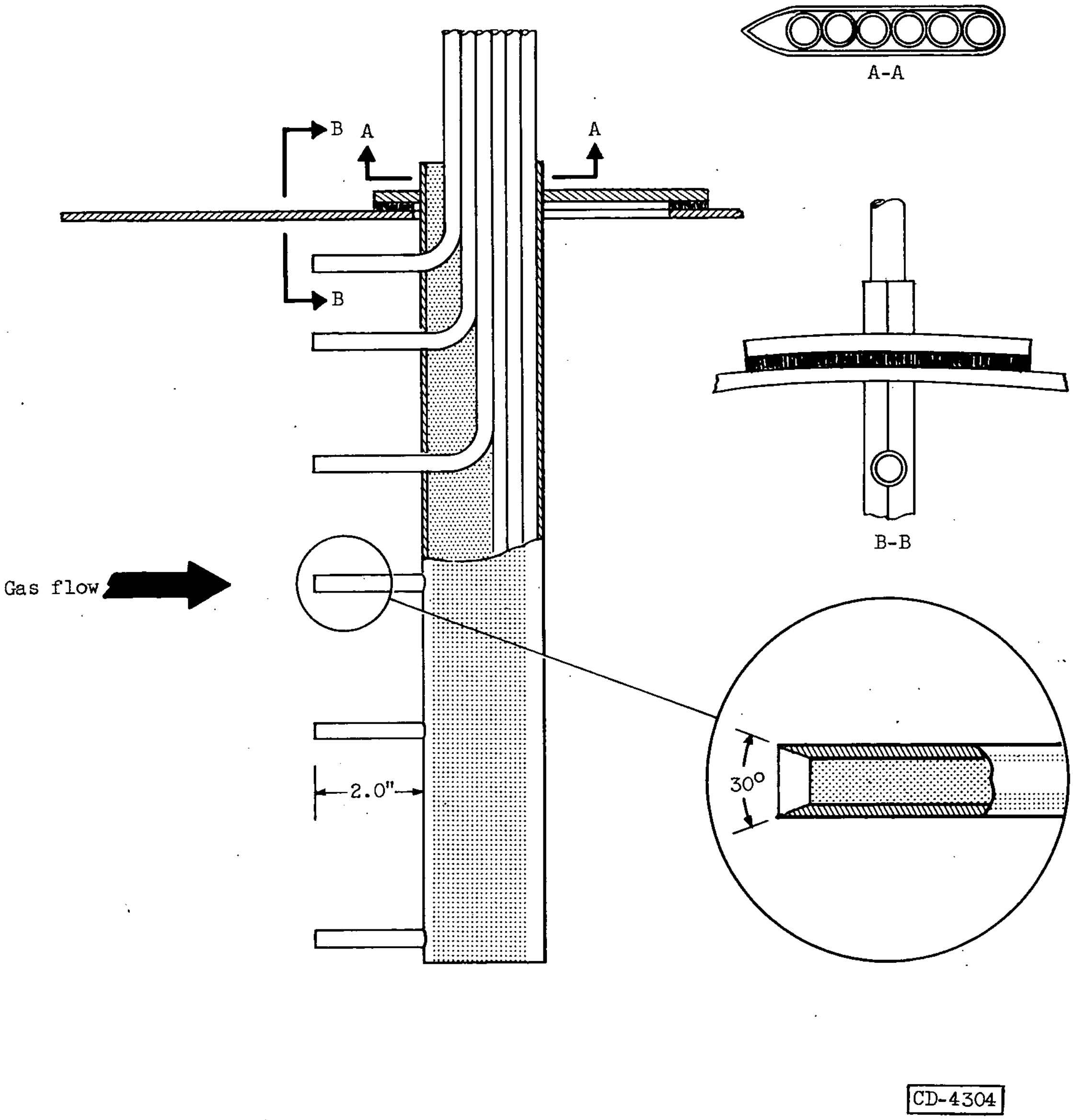


Figure 4. - Typical uncooled exhaust-nozzle-inlet rake for measuring total pressure. (Radial spacing of probes based on equal areas.)

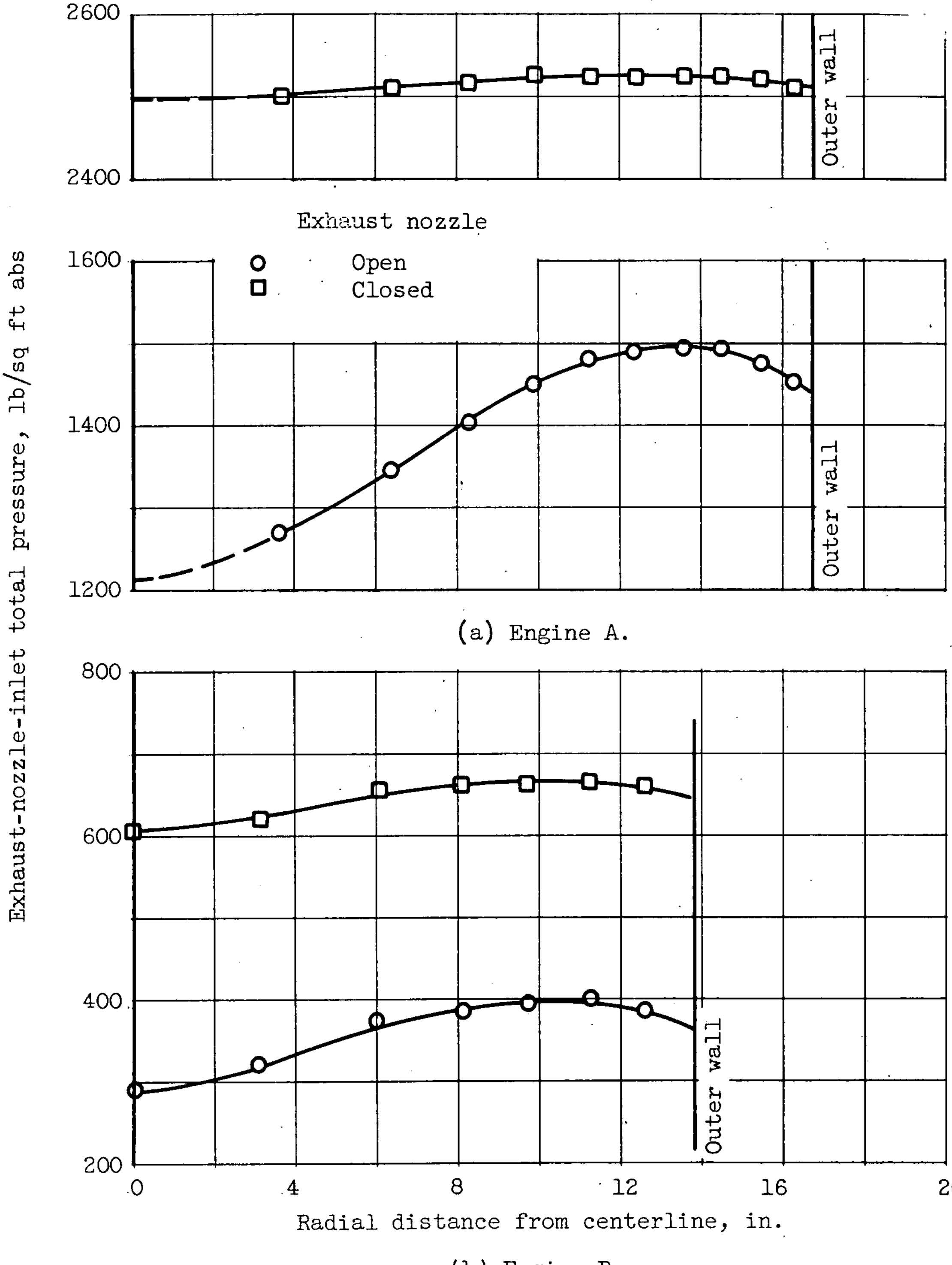


Figure 5. - Typical exhaust-nozzle-inlet total-pressure gradients.

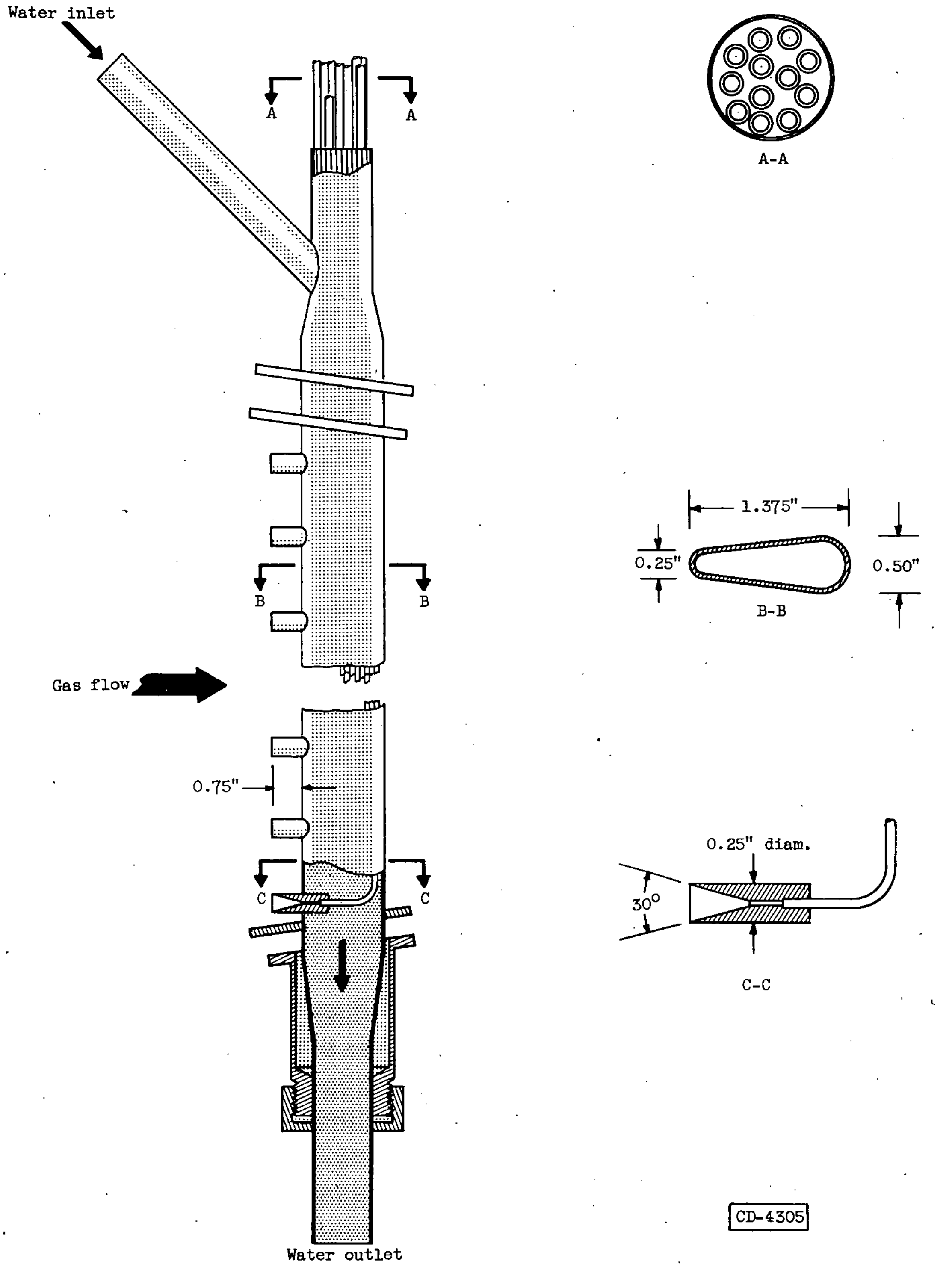


Figure 6. - Typical water-cooled exhaust-nozzle-inlet rake for measuring total pressure. (Radial spacing of probes based on equal areas.)

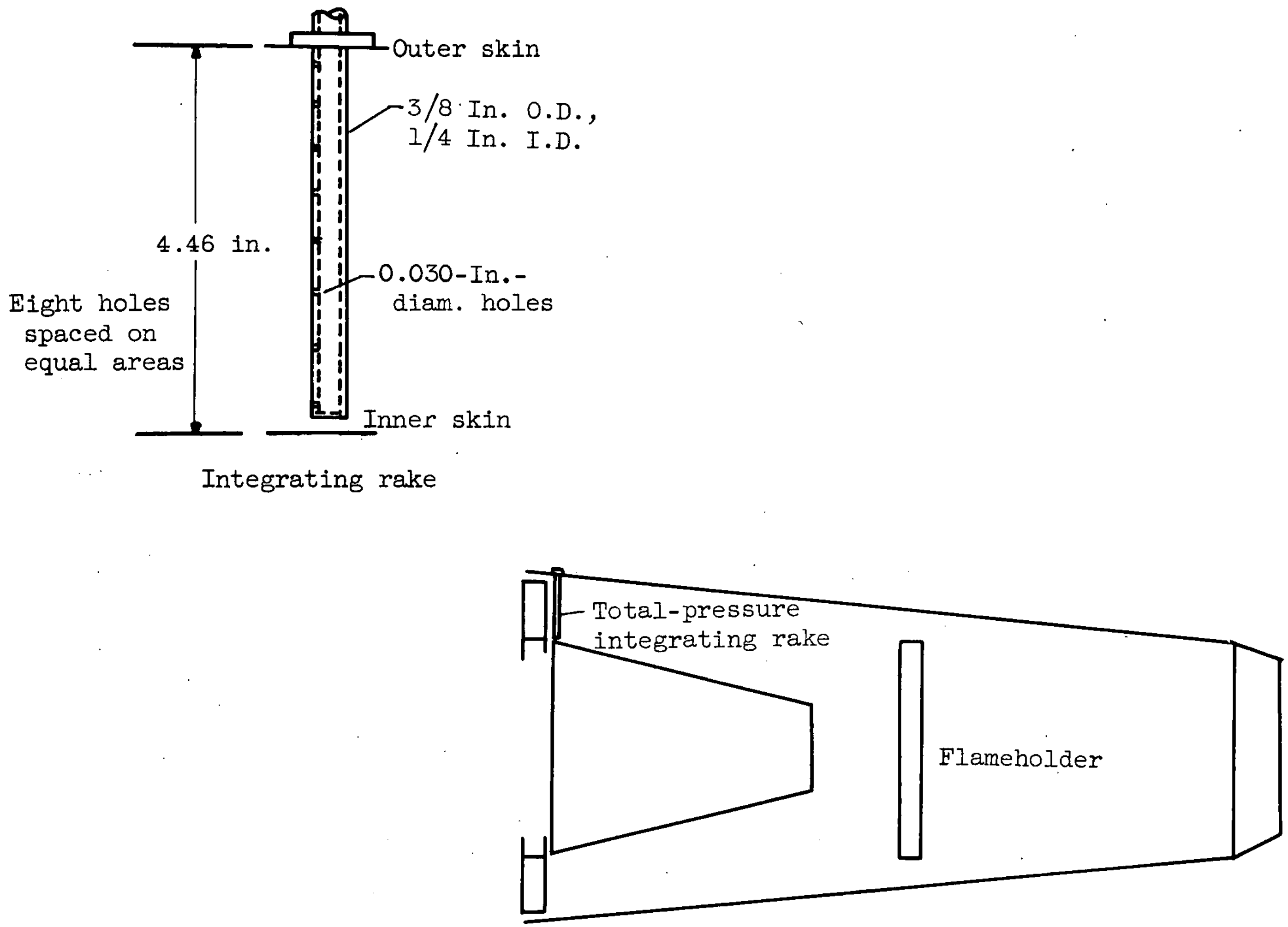


Figure 7. - Design and location of integrating total-pressure probe.

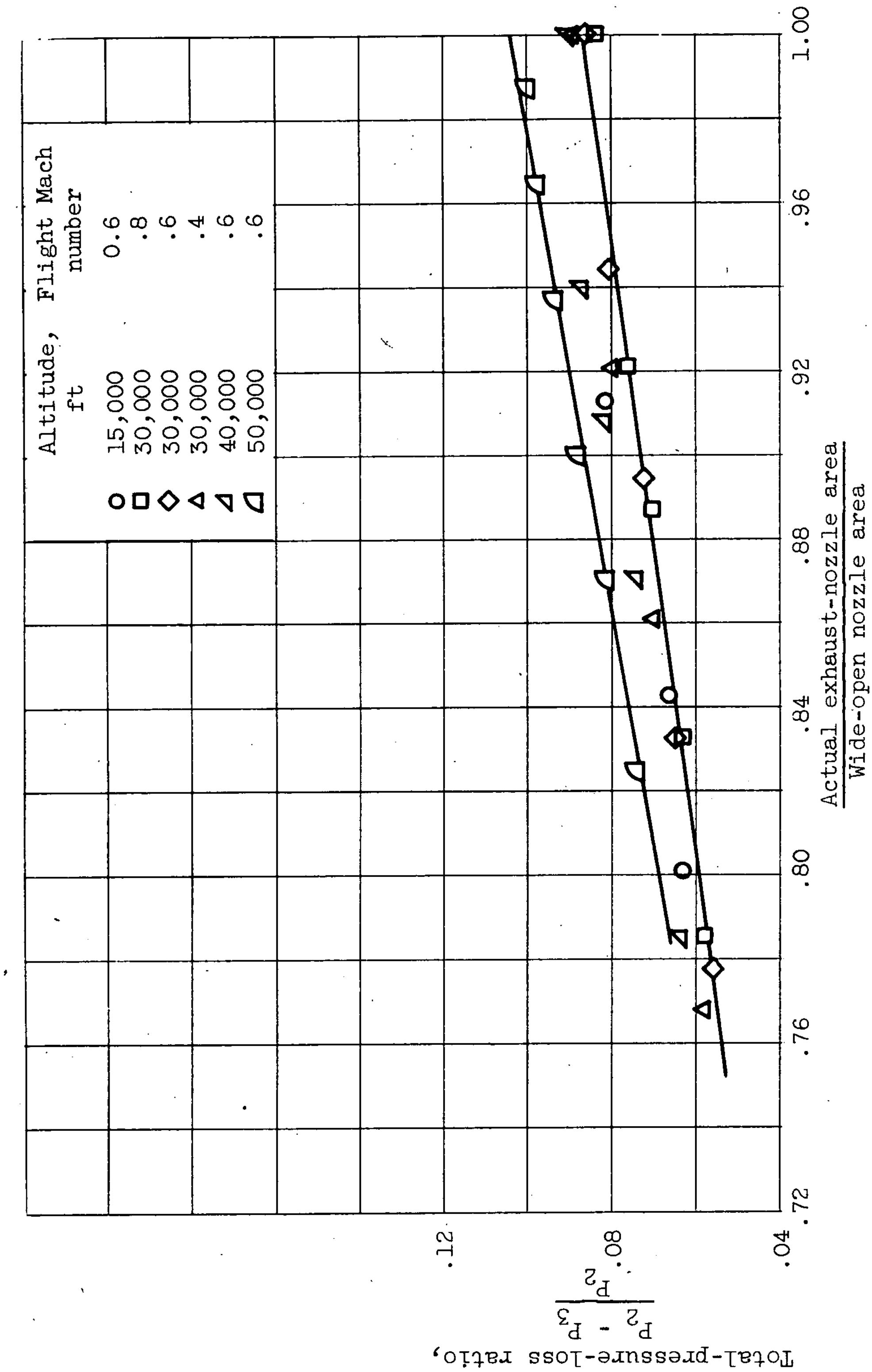


Figure 8. - Typical correlation of afterburner total-pressure loss with exhaust-nozzle area ratio. Maximum engine speed and turbine-outlet temperature.

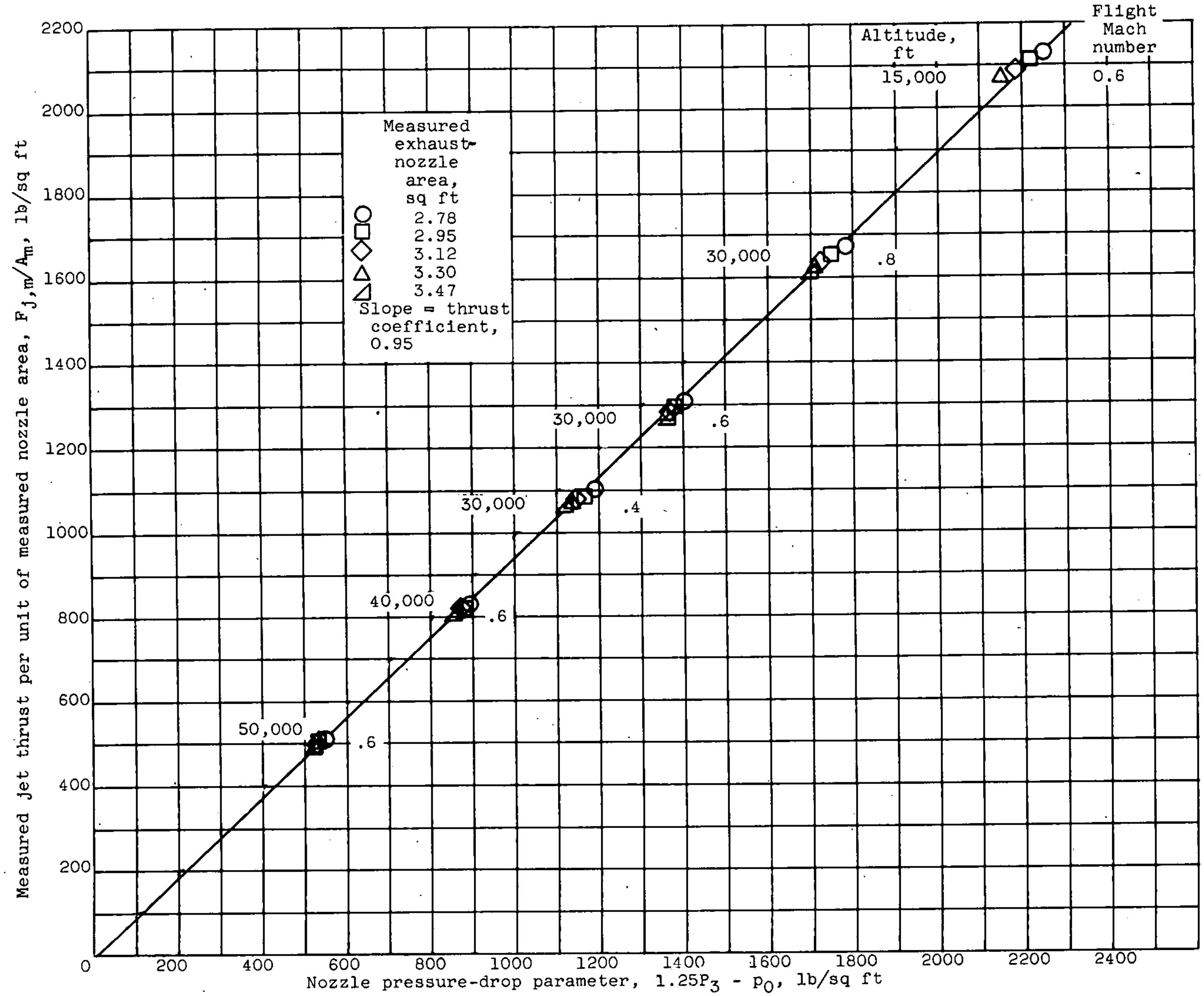
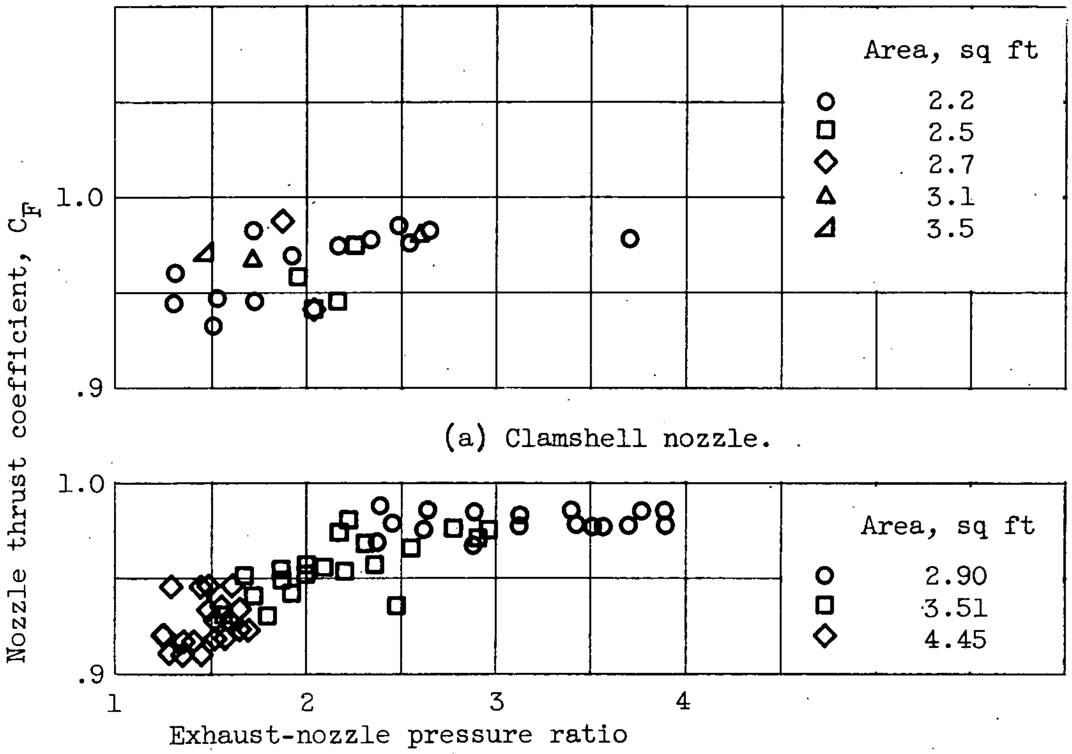
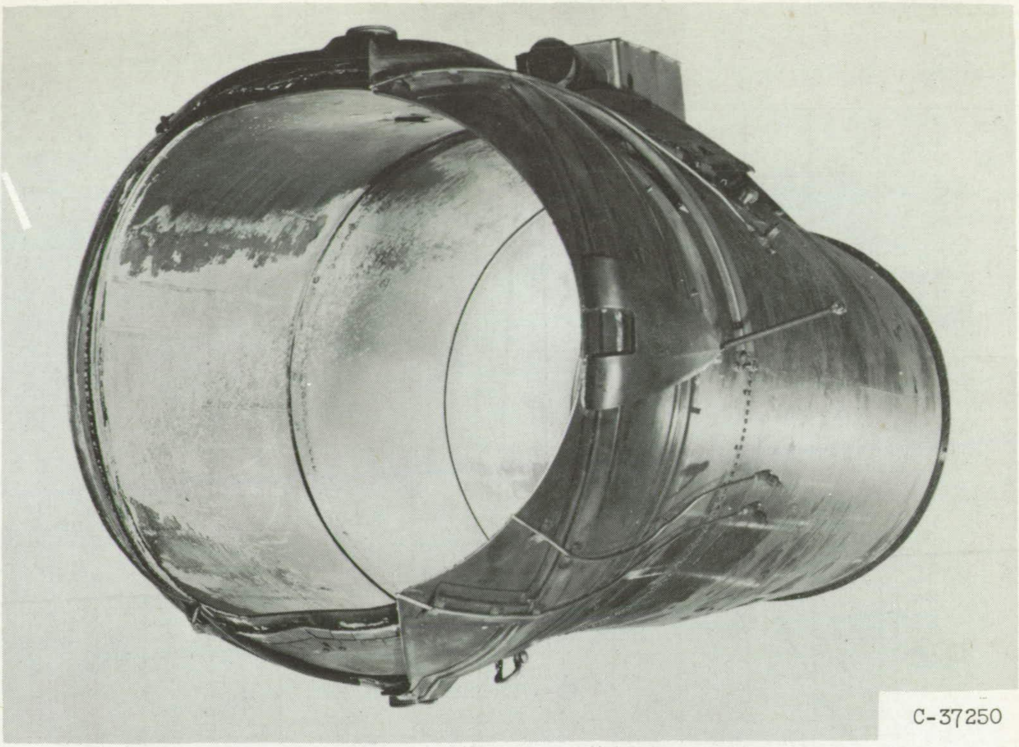


Figure 9. - Correlation of measured jet thrust per unit of measured exhaust-nozzle area for afterburning engine equipped with exhaust nozzle of continuously variable area.



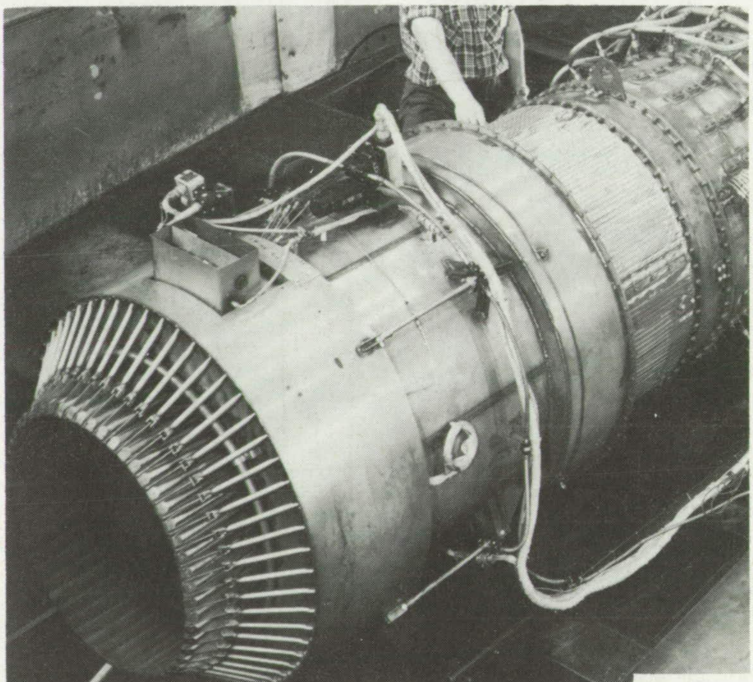
(b) Iris nozzle.

Figure 10. - Variable-area-nozzle thrust coefficient.



C-37250

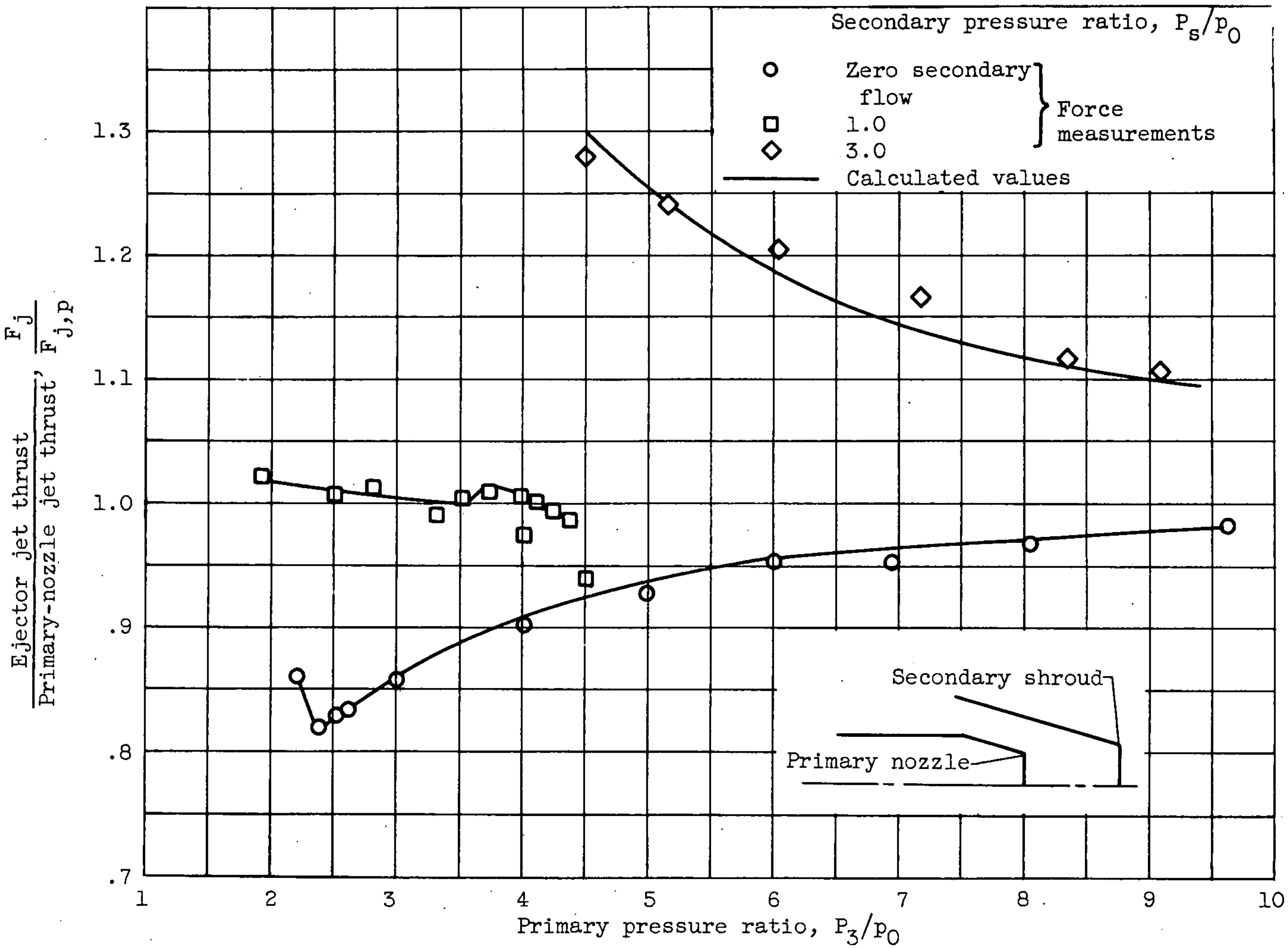
(a) Clamshell nozzle.



C-30514

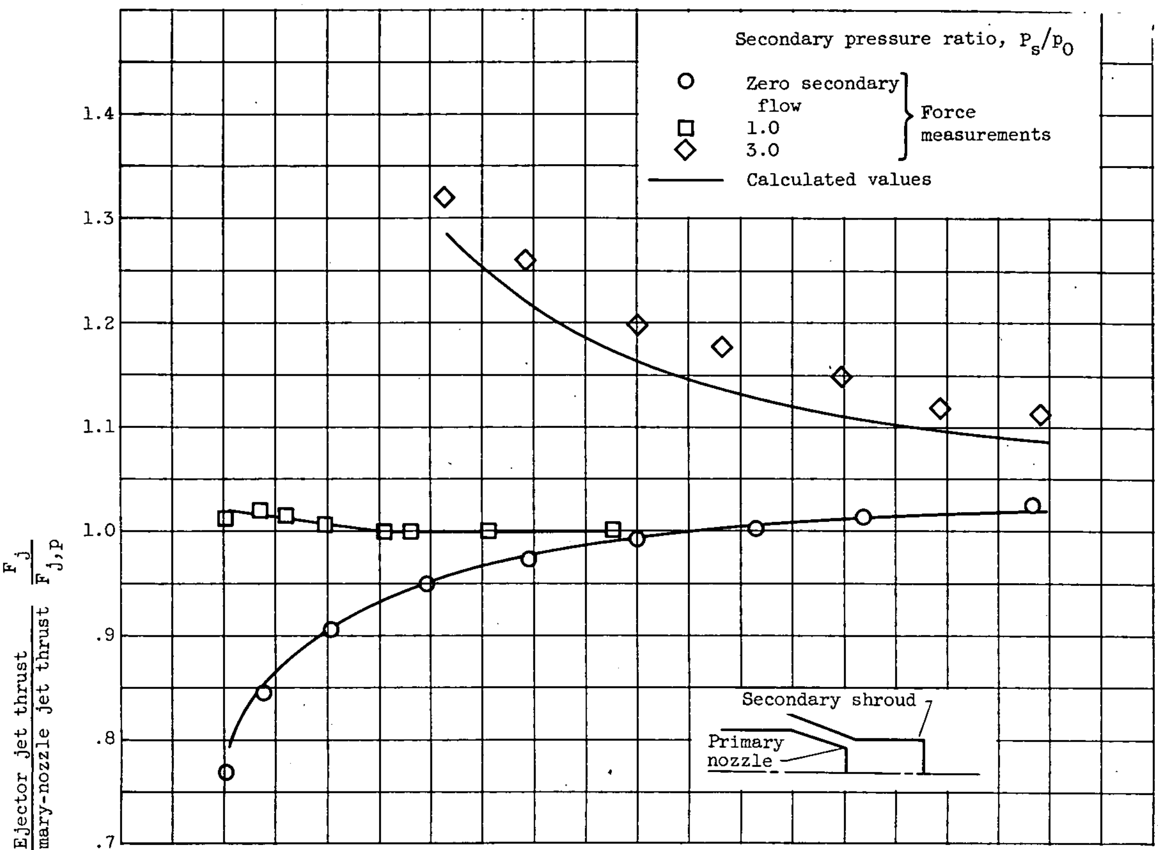
(b) Iris nozzle.

Figure 11. - Variable-area exhaust nozzles.

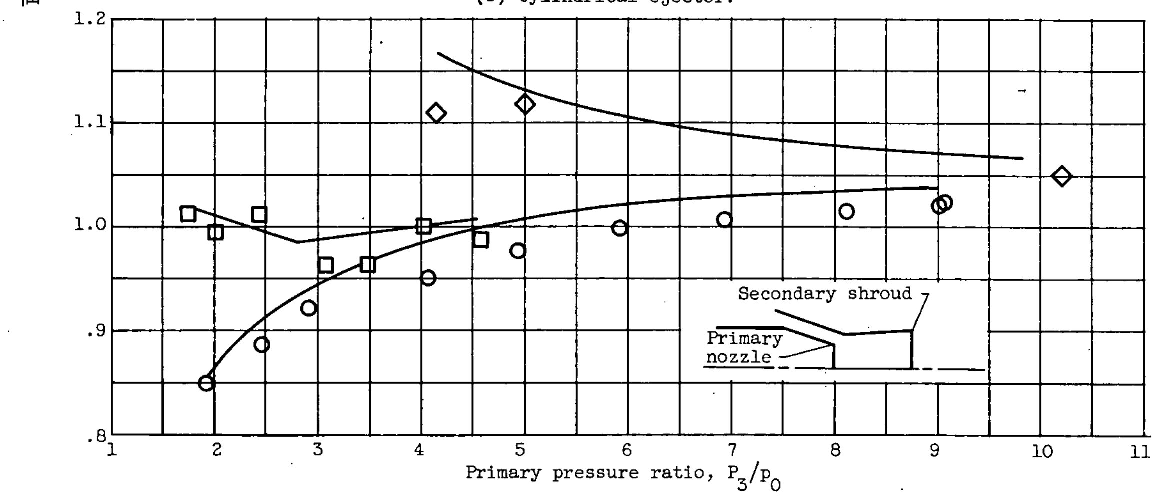


(a) Convergent conical ejector.

Figure 12. - Comparison of calculated and measured jet thrusts for convergent, cylindrical, and divergent ejectors. Exit diameter ratio, 1.2; spacing ratio, 1.6.



(b) Cylindrical ejector.



(c) Divergent conical ejector.

Figure 12. - Concluded. Comparison of calculated and measured jet thrusts for convergent, cylindrical, and divergent ejectors. Exit diameter ratio, 1.2; spacing ratio, 1.6.

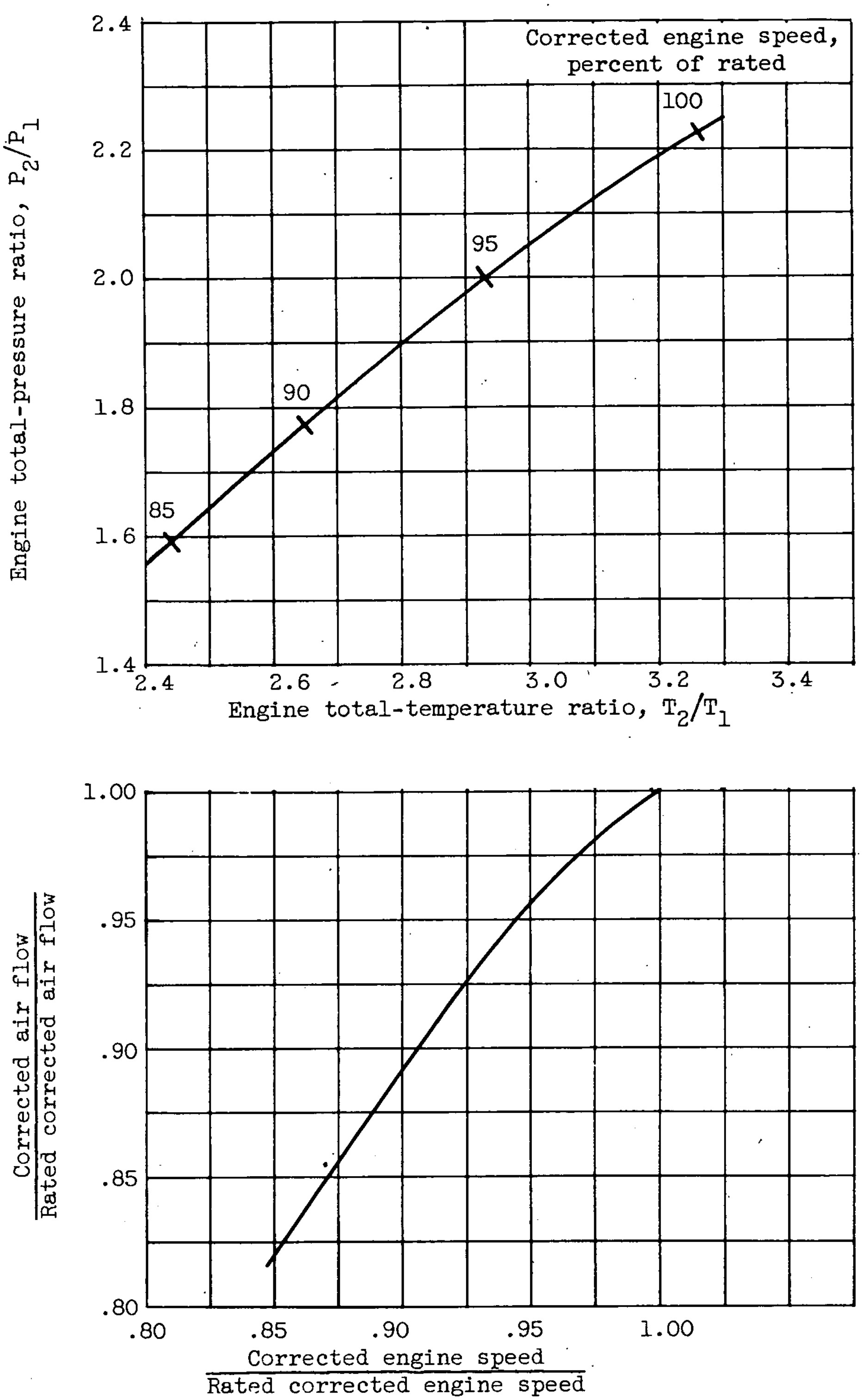


Figure 13. - Typical engine pumping characteristics.

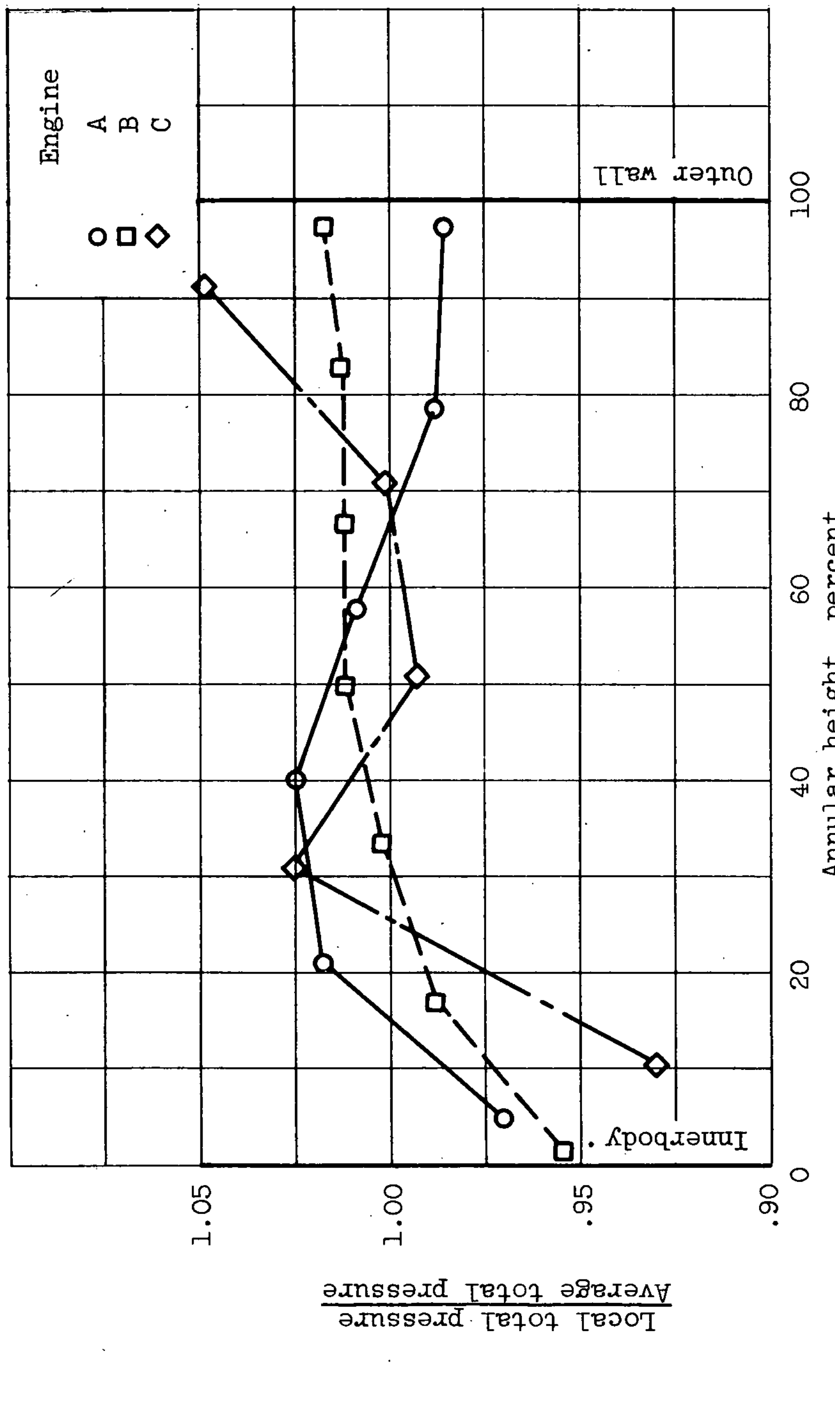
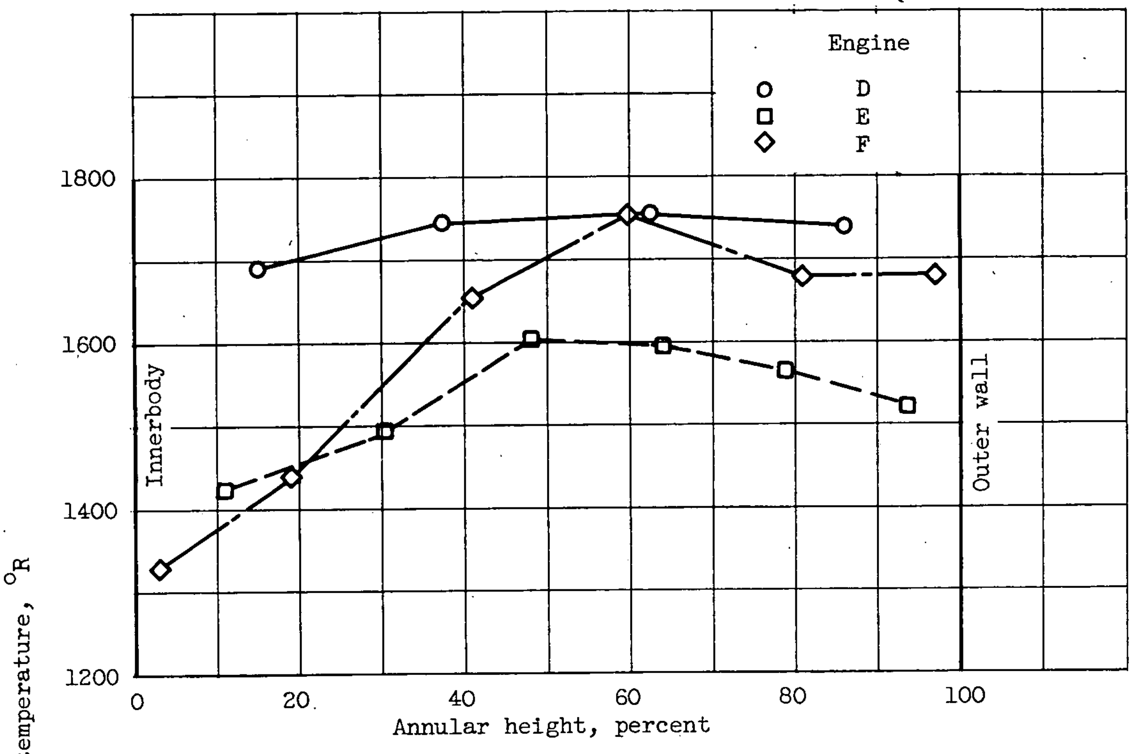
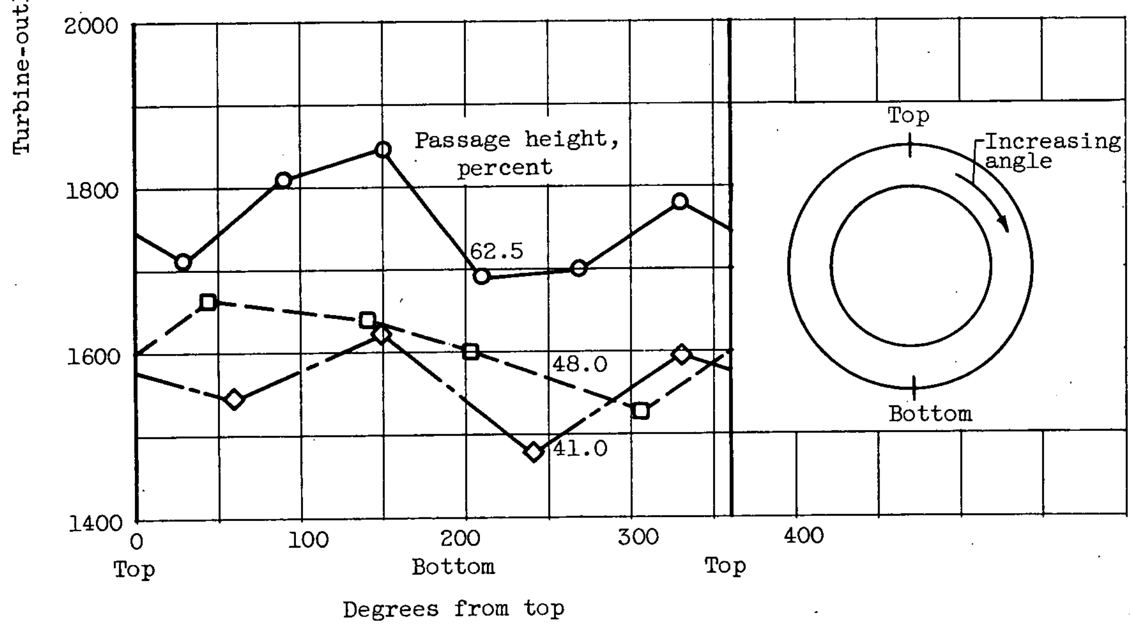


Figure 14. - Radial turbine-outlet total-pressure gradient for three current engines.



(a) Radial gradients.



(b) Circumferential gradients.

Figure 15. - Turbine-outlet temperature gradients for three current engines.